Volume 5 January 1959 No. 1

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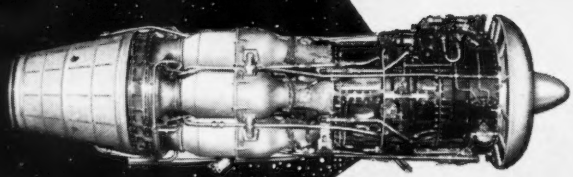
Canadian Aeronautical Journal

CONTENTS

EDITORIAL: LET US NOW PRAISE FAMOUS MEN		1
ULTRA HIGH STRENGTH STEELS	*Dr. W. B. F. Mackay*	3
PILOTING RESEARCH AIRCRAFT	*J. Walker*	13
PERFORMANCE OF A PERFORATED WALL INTAKE AT A MACH NUMBER OF 2.5	*D. Brown*	17
WILL WE BE ABLE TO OPERATE A TURBOJET TRANSPORT ECONOMICALLY?	*C. H. Glenn*	24
C.A.I. LOG Secretary's Letter, Branches, Members, Books, Appointment Notices		31

Published by the
CANADIAN AERONAUTICAL INSTITUTE
Commonwealth Building, 77 Metcalfe Street
OTTAWA

MORE POWER by ORENDA



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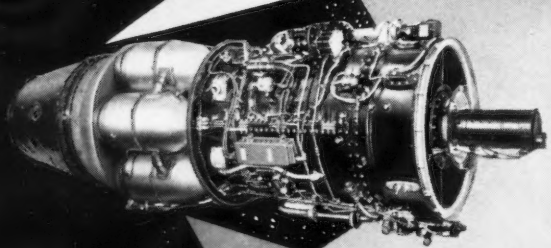

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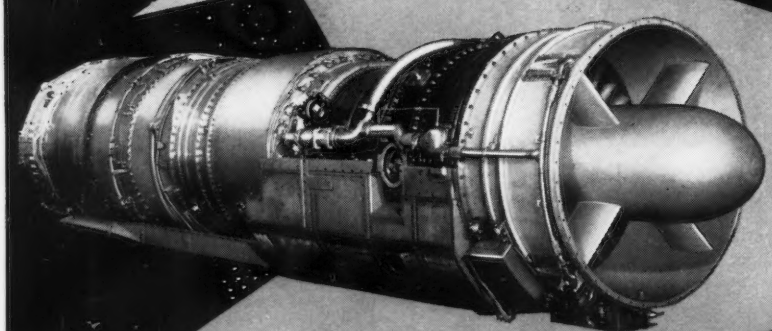
First run on February 10, 1949. More than one million hours of operational service to date in CF-100's and Sabres of the RCAF, West German, Belgian, South African, and Colombian air forces. Thrust-weight ratio, 3:1.



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First run December 15, 1954. First production Iroquois are for the AVRO ARROW Mark II. Thrust-weight ratio, 5:1

Orenda Engines Limited salutes the 50th Anniversary of powered flight in Canada

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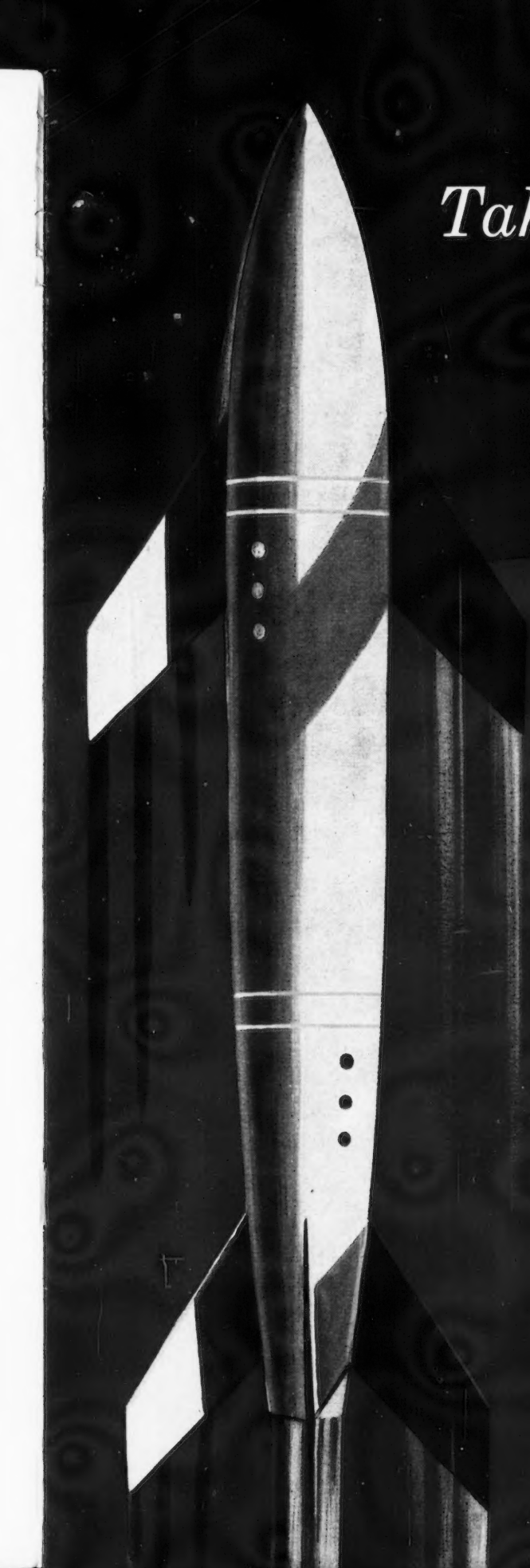

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"Canadair has succeeded in bringing over $200,000,000 of export business to Canada in the past ten years. When you consider that out of every $100,000,000 brought in through export business, $40,000,000 to $45,000,000 goes in wages, not to mention the value of the work to many sub-contractors and suppliers, it is reasonable to conclude that our part in defence business has made a notable contribution to the economic growth of the country."

J. G. Notman, O.B.E., President and General Manager, in an address to visiting Members of Parliament, July, 1958.

The two aircraft pictured above, the "540" and the "CL-44", are in production today at Canadair for the Royal Canadian Air Force: one is for carrying personnel over short-to-medium ranges at high speed, the other for carrying personnel and cargo over very long distances at very high speed.

These two airplanes possess outstanding characteristics in their respective roles, and have excellent possibilities of being sold abroad. Such export sales, if realized, will make a notable contribution to our economy and reduce materially the unit costs of aircraft produced for the Air Force.

As a consequence, Canadair is making every possible effort to sell the "540" and the "CL-44" to other governments and to national and international airlines. Sales teams are hard at work in many parts of the world providing technical information, operating characteristics and costs, and all other pertinent details required by potential customers.

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Subscription—$4.00 a year. Single copy—50 cents.

Published monthly, except in July and August.

The Institute is not responsible for statements or opinions expressed in papers or discussions printed in its publications.

All communications should be addressed to The Secretary, Canadian Aeronautical Institute, 77 Metcalfe St., Ottawa 4, Ontario, Canada.

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EDITORIAL

LET US NOW PRAISE FAMOUS MEN

THE year 1959 is the 50th anniversary of the flight of the Silver Dart and it is appropriate at this time to review some of the related facts and dates.

On the 23rd February, 1909, J. A. D. McCurdy flew the Silver Dart from the ice of Baddeck Bay, Bras d'Or Lakes in Cape Breton Island, for a distance of over half a mile. This was the first flight of a powered heavier-than-air machine in Canada and, in fact, the first such flight in the British Empire by a British subject.

The aircraft was built at Hammondsport, N.Y., by the Aerial Experiment Association, a group headed by Dr. Alexander Graham Bell. The other members of the Association were Glenn H. Curtiss, an American engine expert, Lt. T. E. Selfridge of the U.S. Army, and two Canadian engineers, J. A. D. McCurdy and F. W. (Casey) Baldwin. Dr. Bell, though at that time a U.S. citizen and normally living in the States, had a summer home at Baddeck; Curtiss' engine shop was at Hammondsport and it was there that the Association's aircraft were built and given their initial tests.

The first Canadian to fly was Baldwin, who flew the Association's first aircraft, the Red Wing, on Lake Keuka near Hammondsport on the 12th March, 1908.

The Silver Dart made a number of successful flights, including a closed circuit of several miles at Baddeck on the 10th March, 1909. It was finally destroyed in a crash at Petawawa, not far from Ottawa, on the 2nd August, 1909, during demonstrations to the Army. Its engine survived and, after an eventful career, which included a long period of immersion following the sinking of a launch in which it was installed, it now stands in the Museum of the National Research Council in Ottawa.

There has been a great deal of debate on both sides of the Atlantic about who was the first to do what. Though there are indeed some rival claims, it is generally accepted that the first powered flight in a heavier-than-air machine was made by Orville Wright at Kitty Hawk on the 17th December, 1903. The subsequent history of those early days is clouded by the view, widely held at the time, that a flight which was not completely successful from takeoff to landing should not count; moreover some of the hops made by A. V. Roe in the summer of 1908 are still subject to argument, though they are not generally accepted as true flights.

The first flight in England is credited to Col. S. F. Cody. His flight of 1,390 ft on the 16th October, 1908, admittedly ended in a crash landing, due to an attempt to avoid some trees, but he flew again, this time completely successfully, on the 2nd January, 1909. Cody was a U.S. citizen at the time.

The first flight by an Englishman was made by Henry Farman in France in October, 1907. Everybody regarded Farman as a Frenchman but, in fact, he was a British subject at the time – and remained so until 1937. (Incidentally, Farman died only last year at the age of 84.)

The flight of the Silver Dart, piloted by McCurdy, on the 23rd February, 1909, was therefore neither the first flight in the British Empire (Cody predated it by at least 7 weeks) nor the first flight by a British Subject (Farman predated it, as indeed did Baldwin). But it is safe to say that it was the first flight by a British Subject in the British Empire, and certainly the first flight in Canada.

In 1959, as space itself lies before us, it is well that we should look back and honour the men who, fifty years ago, first set our feet on the pathway to the stars.

Celebrating the First Powered Flight in Canada, 23rd February, 1909

SPECIAL ANNIVERSARY MEETING

THE QUEEN ELIZABETH

MONTREAL

23rd and 24th February, 1959

TECHNICAL SESSIONS – The programme will look towards the future, with sessions on Atmospheric Flight, Space Flight, Satellites and Propulsion. Each session will be addressed by an eminent scientist or engineer from abroad–from Great Britain or the U.S.A. – as well as by a distinguished Canadian.

THE DINNER – Representatives of the Aerial Experiment Association (of 1909), including the Hon. J. A. D. McCurdy who made the historic flight, will be among the guests at the Dinner on the 23rd February.

His Excellency The Rt. Hon. Vincent Massey, Governor-General of Canada, will be the Guest of Honour and Principal Speaker.

LUNCHEON – In keeping with the theme of the technical programme, a Luncheon sponsored jointly by the Astronautics and Propulsion Sections will be held on the 24th February and will be addressed by the Director of the David Dunlap Observatory, University of Toronto.

ANNUAL GENERAL MEETING

The Annual General Meeting of the Institute

will be held at

KELTIC LODGE

INGONISH BEACH, N.S.

on the

15th, 16th and 17th June, 1959

The Programme, which is now being prepared, will include Sessions on

Flying — Past, Present and Future,

and Hydrodynamics

as well as the annual Business Meetings of the Institute

and the Specialist Sections.

This meeting affords an opportunity for the presentation of papers by members of the C.A.I. The Council is most anxious to encourage Canadian papers and hopes that any member wishing to contribute to either of the above-mentioned Sessions will submit a summary of his paper for consideration. Such summaries must be in the hands of the Secretary by the 31st January 1959.

ULTRA HIGH STRENGTH STEELS†

by Dr. W. B. F. Mackay*

Atlas Steels Limited

INTRODUCTION

ULTRA high strength steels are not new to the tool and specialty steel manufacturer. For half a century, this industry has produced steels, such as the high speed grades, which have gone into service at hardness levels up to 65/67 Rockwell "C" and with ultimate tensile strengths in excess of 400,000 psi. The tool and die industry is away ahead of the aircraft industry if we consider the use of steel from a purely high strength point of view. This gives rise to the question, "What is meant by ultra high strength steels?" The steels we are interested in for the aircraft industry are difficult to define or categorize due to the range of compositions and properties which they represent. For the purpose of this paper, the term "'ultra high strength steels" is defined as "those high cleanliness aircraft steels, stainless or otherwise, which possess ultimate tensile strengths varying from 200,000 to 300,000 psi, have widely differing load carrying ability over the temperature range −85° to 1,000°F, and whose toughness and ductility are usually borderline according to presently accepted standards".

The basic problem is therefore not one of strength, but rather one of utilizing a fraction of the strength now available and combining it with various other desirable characteristics so that the steel may be fabricated and enter service with properties acceptable to the aircraft designers' present thinking.

HISTORICAL BACKGROUND

The Ni-Cr-Mo low alloy type steels, such as SAE 4340, have been used by the aircraft industry for many years. While aluminum alloys of the 2024 class with a strength in the neighbourhood of 60,000 psi were used, SAE 4340 heat treated to 180,000 psi was considered adequate. The introduction of stronger aluminum alloys of the 7075 type and the advent of titanium alloys created a demand for higher strength steels in order that they might maintain a competitive strength/weight ratio. To attain a comparable, or even superior, strength/weight ratio to that offered by competitive materials, it was then necessary to place steel parts in service with strengths in excess of 220,000 psi.

Two approaches towards higher strength were possible: firstly, the tempering temperature could be lowered and, secondly, the carbon content could be reduced and the alloy content increased. Lockheed[1] pioneered the first approach in raising the strength of standard 4340 to the 260,000-280,000 psi range by decreasing the tempering temperature from above 800°F down to the 400/450°F range. This low tempering treatment skipped the 500°F embrittlement range. Up to this time, the aircraft industry generally was strongly opposed to tempering temperatures below 800°F for fear of undue brittleness. The resulting lowered impact figures obtained from conventional tests at these high strength levels were not considered by Lockheed as sufficient grounds to preclude its use in applications where service loadings were not considered analogous to the impact test. A second and more gradual approach to higher strengths was taken by other concerns. A series of alloys were developed which, by reducing the carbon content from approximately 0.40% to about 0.30%, and increasing the alloy content (notably molybdenum and vanadium), strength levels from 220,000-240,000 up to 260,000-280,000 psi were obtained with little or no change in toughness or ductility. The Republic-Bendix AMS-6427[2] steel is an excellent example of this approach. A number of older British steels, for example EN30B, would also fit into this category.

A slightly different line of attack utilizing higher silicon contents resulted in Crucible's well known Hy-Tuff[3] and similar steels. Silicon has the effect of shifting the embrittlement range to the right on an increasing temperature scale (Figure 1) and increasing the resistance to softening on tempering[4, 5, 6]. This permits tempering at 600°F to approximately the same strength as a 400°F temper would produce in a steel of normal silicon content.

The higher 600°F tempering temperature as opposed to the 400°F temperature gives additional relief of residual quenching stresses. This results in an appreciable increase in the yield/tensile ratio due to a straightening of the stress strain curve[6].

Steels such as 4340, AMS-6427, Hy-Tuff, and HS-220 cover the field up to and including the 260,000-280,000 psi range. For strengths in the 300,000 psi range, certain super grades have evolved like Super Hy-Tuff, Tricent, and Super Tricent. Generally speaking, these steels take advantage of a combination of the salient features utilized in the previously mentioned grades, plus the retention of 0.40% or more carbon. That is, they have their alloy content increased, utilize high silicon contents, and are tempered below the embrittlement valley[4].

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†Paper read at the Joint I.A.S./C.A.I. Meeting in Ottawa on the 8th October, 1958.

*Research and Development Engineer.

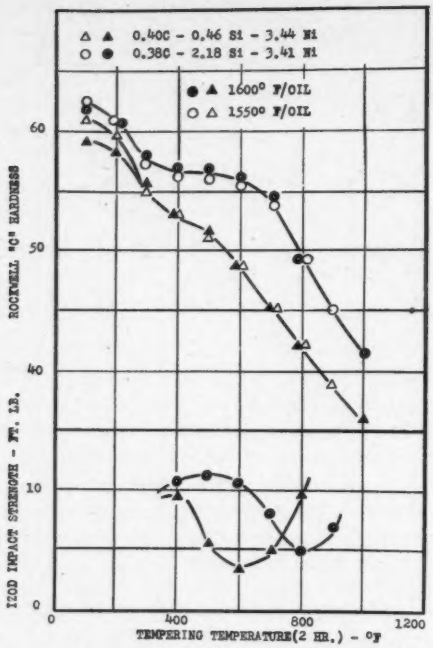
△ ▲ 0.40C - 0.46 Si - 3.44 Ni
○ ● 0.38C - 2.18 Si - 3.41 Ni
● ▲ 1600° F/OIL
○ △ 1550° F/OIL
ROCKWELL "C" HARDNESS
IZOD IMPACT STRENGTH - FT. LB.
TEMPERING TEMPERATURE(2 HR.) - °F
ASM Transactions

Figure 1
Effect of silicon content on hardness and impact strength of 3% nickel steels, refrigerated after quenching[5]

As the low alloy constructional steels were being developed to very high strengths at moderate temperatures, other requirements, which were to restrict their usefulness, loomed larger on the horizon. A major problem has been created due to aerodynamic heating in newer types of high performance aircraft and other vehicles. In addition, the propulsion units of these vehicles not only demand higher temperature resistant properties in their own construction, but also in the surrounding airframe components. This demand for good load carrying ability at moderate temperatures has extended the use of steel into new areas like skins and sub-surface components, which were once completely dominated by the light alloys. This situation has given a major role in aircraft construction to the stainless steels and the hot work tool steels.

PRINCIPAL CATEGORIES OF CURRENT ULTRA HIGH STRENGTH STEELS

The ultra high strength steels developed for present-day use may be categorized under three broad headings. These are:

(1) Low alloy construction steels,
(2) Stainless steels, and
(3) Hot work tool steels.

The following is a further breakdown of the above categories with a brief mention of some of the main advantages and disadvantages inherent in these individual alloy groups.

Low alloy constructional steels

These alloy steels (Table 1) are relatively inexpensive due to their low and non-strategic alloy content. They are capable of developing strengths up to 300,000 psi at a considerable sacrifice of their excellent lower strength

TABLE 1
CHEMICAL COMPOSITION OF SOME ULTRA HIGH STRENGTH CONSTRUCTIONAL STEELS

Designation	A.M.S. No.	Typical Chemical Composition %						
		C	Mn	Si	Ni	Cr	Mo	V
*4340	6415	0.39	0.72	0.28	1.92	0.79	0.24	
4340 Mod	6427	0.30	0.90	0.25	1.83	0.85	0.43	0.08
*Ultimo No. 4		0.40	0.61	0.25	1.66	0.66	0.34	
*Dowcan 110 D		0.37	0.75	0.31	1.80	0.82	0.33	0.04
*EN30B (AHT-28)		0.30	0.50	0.27	4.10	1.24	0.25	
H.T. 220	6407	0.30	0.70	0.55	2.05	1.20	0.45	
*Hy-Tuf	6418	0.29	1.42	1.45	1.78	0.12	0.37	
4350		0.50	0.75	0.25	1.83	0.80	0.25	
HS-260		0.40	0.85	0.60	2.20	1.45	0.50	
Super Hy-Tuf		0.40	1.30	2.30	–	1.40	0.35	0.20
Hi C Super Hy-Tuf		0.47	1.28	2.42	–	1.11	0.42	0.25
98B40 Mod		0.40	0.75	0.25	0.85	0.80	0.20	Boron
U.S.S. Strux		0.43	0.90	0.55	0.75	0.90	0.55	Boron
Tricent (300M)		0.40	0.80	1.60	1.83	0.85	0.38	0.08
Super Tricent		0.55	0.80	2.10	3.60	0.90	0.50	

*Actual Atlas Heat Analysis, balance from Reference 4.

ductility and toughness[4]. Good impact resistance at moderate strength levels is obtained down to low temperatures with many of these steels. The principal disadvantages they possess are poor elevated temperature properties which limit their use to approximately 400°F, lack of corrosion resistance, and high residual stress. Certain processing difficulties are also encountered, not all of which are a particular affliction of this group alone. Some of these are: a tendency towards distortion during the quench, susceptibility towards decarburization, and grinding cracks, and problems in welding and machining[7]. For components in subsonic aircraft, however, they have and will continue to play an important role.

Stainless steels

Austenitic stainless steels

The non heat treatable grades of stainless steels have been used in aircraft construction for a number of years. Although the basic 18-8 chromium-nickel Type 302 does not fully satisfy our definition of an ultra high strength steel, it is interesting to note that over twenty-five years ago this grade was used almost exclusively in the all-stainless amphibian "Seabird", built by Fleetwings.

Type 301 (17-7) develops the highest strength of all the work hardenable austenitic steels[8]. Ultimate tensile strengths up to, and in excess of, 200,000 psi are obtained in the extra hard condition at about 45% cold reduction. The chemical composition of Type 301 with 17% chromium and 7% nickel makes it the most metastable grade of the Type 300 austenitic stainless steels. Its high strength is due to two sources: firstly, its inherently high work hardening characteristics of the austenitic matrix and, secondly, as the result of a partial transformation of austenite to martensite during cold working. This high strength is accompanied by relatively high ductility and excellent corrosion and oxidation resistance. Due to the necessity of obtaining strength through a certain degree of cold reduction, its formability suffers. This restricts its application at high hardness to components requiring only mild-forming operations. The tear strength of this grade is quite good. One objection levelled at Type 301 is that it possesses a relatively low compressive yield strength due to the directionality in mechanical properties imparted by cold reduction. Usually, with proper design, Type 301 structures may be produced with good compressive load carrying values[9]. Another approach is a stress relieving treatment at 800/900°F which can eliminate directionality and improve the compressive yield strength. The elevated temperature properties of Type 301 compare very favourably with the heat treatable and precipitation hardening grades when subject to prolonged heating up to 800°F.

Martensitic stainless steels

Except for small parts, the 12-13% chromium grades have not been used extensively in airframes. One of the exceptions to this statement and a classic example of an early application was its use for 55% by weight of the primary structure in the British airship R-101.

These grades may be heat treated to high strength levels, but certain production problems have been largely responsible for restricting their use. Some of these difficulties are[7]: susceptibility to decarburization, machining and welding problems, and distortion in heat treatment. The oxidation resistance of these grades is excellent, and their corrosion resistance in the hardened condition good, but not comparable with the austenitic grades. The elevated temperature strength of standard Type 420 deteriorates rapidly beyond 800°F. The tear strength and ductility is lower than for the other stainless groups. Stress corrosion cracking has been reported at high strength levels with this alloy. If proper design, fabrication techniques and heat treatment are employed, this should not place it in a class apart from other grades.

Type 410 is included, although its maximum tensile strength barely exceeds 200,000 psi. It has been used for various small fittings and as thin strip for compressor blading. Mention should be made of the relatively new U.S.S. 12-MoV steel. This modified Type 410 alloy, with a strength of 225,000 psi, would place it in the ultra high strength classification. Type 422 and 422 (modified) are hardenable grades used at temperatures approaching 1,200°F. This grade is not produced in sheet form and, compared with the straight chromium grades, its corrosion resistance suffers on account of the additional elements added[10]. Although Type 422 (modified) at 240,000 psi and a few other 400 series alloys, including 414, 418, 419 and 431, might be classed as high strength steels, they are more often regarded as special stainless steels or low temperature superalloys. Standard AISI Type 440 is not included, as it is restricted in use due to lack of ductility and toughness in comparison with Type 420.

Precipitation hardening stainless steels

This relatively new category of stainless steels may be sub-divided into the three classes shown in Table 2.

Martensitic PH stainless steels

The martensitic class of alloys possesses a balanced composition such that, on quenching from the solution annealing temperature, the austenitic phase transforms to a low carbon martensite with some delta ferrite. On tempering, precipitation occurs within the tempered martensite (alpha ferrite) matrix, thus increasing its hardness and strength. The formability of these alloys appears to be limited in comparison with other PH stainless steels due to their "as quenched" martensitic-type structure. This class of alloy is not available as sheet or strip, but it finds some use as forgings, bars, plates and castings. The rapid rate of work hardening in 17-4 PH would further restrict the use of this grade in sheet form[11]. Although Stainless W is probably the earliest PH stainless alloy to be exploited commercially, the later 17-4 PH with improved weldability has achieved wider acceptance.

Semi-Austenitic PH stainless steels

The semi-austenitic stainless steels have an unbalanced chemical composition so adjusted that in the annealed condition the material is fully austenitic, but, by suitable heat treatment, the austenite can be made to transform to martensite. Strength may be obtained by a double aging treatment, or by a refrigeration treatment, the latter being the most commonly used. In the fully

TABLE 2
CHEMICAL COMPOSITION OF SOME HIGH STRENGTH STAINLESS STEELS

Type		Designation	Typical Chemical Composition %									
			C	Cr	Ni	Mn	Si	Mo	Cu	Al	Ti	Other
Austenitic		*Type 301	0.09	17.42	7.03	0.80	0.65	0.24				
Martensitic		*Type 410	0.13	12.70	0.16	0.45	0.53	0.04				
		*Type 420	0.35	13.52	0.27	0.42	0.33	0.30				
		Type 422	0.20	13.00	0.75	0.75	0.35	1.00				V 0.30 W 1.00
Precipitation Hardening	Martensitic	Stainless W	0.07	16.69	7.12	0.55	0.52			0.30	0.76	
		17–4 PH	0.04	16.50	4.0	0.60	0.40		4.0			Cb + Ta 0.35
		V2B	0.07	10.0	19.0			3.0	2.0			Be 0.15
	Semi-Austenitic	17–7PH	0.07	17.0	7.0	0.60	0.40			1.15		
		15–7 Mo	0.07	15.0	7.0	0.60	0.40	2.50		1.0		
		AM 350	0.08	17.0	4.2	0.60	0.40	2.75				N 0.10
		AM 355	0.13	15.44	4.35	0.95	0.40	2.74				N 0.10
		FV 520	0.07	16.0	6.0			1.5	1.5		0.3	
	Austenitic	17–10 P	0.12	17.0	10.25							P 0.25
		17–14 Cu Mo	0.12	16.0	14.0			2.5	3.0		0.25	Cb 0.45
		HNM	0.30	19.0	9.5	3.5						P 0.30

*Actual Atlas Heat Analysis, balance from Manufacturers' literature.

annealed condition, the structure of these alloys consists of delta ferrite (5-20%) in an austenitic matrix. The austenite is fairly stable in this state and welding and severe forming operations may be carried out. The refrigeration treatment is carried out in three stages: a conditioning or "Trigger" anneal at 1,700/1,750°F, cooling to −100°F and then aging at 850/950°F followed by air cooling. The solutionizing temperature of the conditioning anneal being some 200°F lower than the full annealing temperature creates an unstable austenite which is very prone towards transforming to martensite. This unstable austenite is transformed to martensite by the sub-zero treatment. Aging results in increased strength by precipitation within the tempered martensite matrix. The double aging treatment is basically a high aging treatment (1,375/1,400°F), cooling to room temperature or slightly below and then re-aging at a lower temperature (850/1,050°F) followed by air cooling. The first aging treatment raises the Ms point by precipitating carbides out of the austenitic matrix, which has become unstable. Cooling to 50/60°F within a one-hour interval transforms the austenite to martensite. The second aging tempers this martensite while precipitation occurs throughout the structure. Maximum strength is obviously obtained from the refrigeration treatment, as the austenite prior to the martensitic transformation has a higher carbon content due to the avoidance of carbide precipitation, as in the case of the double aging treatment.

The semi-austenitic AM-350 and AM-355 are not considered to be true PH alloys because they do not rely on precipitation hardening, a fact which results in their room temperature properties being lower than 17-7 PH or 15-7 Mo. Due to their heat treatment and use, however, they are always classified in the semi-austenitic PH group. Actually, an increase of approximately 2 Rockwell "C" is attributed to a precipitation reaction during tempering.

The aircraft industry has accepted the semi-austenitic alloys to a greater extent than any of the other PH classes. This has been due principally to their combination of good formability and good weldability with reasonable strength and corrosion resistance. Compared with other PH classes, the semi-austenitic steels have superior strength, weldability, formability, best load carrying ability up to 800°F, and fair corrosion resistance.

Although most of the ultra high strength steels are superior to the semi-austenitic PH grades in strength at room and elevated temperatures and the austenitic stainless steels possess better corrosion resistance, their all-around good formability and the relative ease with which they can be processed to thin, practically decarb-free sheet has established a definite place for them in aircraft construction[7, 11].

Austenite PH stainless steels

The highly alloyed austenitic alloys are balanced so that a stable austenite plus possibly some delta ferrite is retained after quenching from the solutionizing temperature. Hardening is effected by aging in the 1,000/1,300°F temperature range. This treatment results in the precipitation of complex carbides, phosphides etc in the austenitic matrix. While the solution annealed condition gives good ductility, the maximum strength attainable is not high. The weldability of these alloys as a group is not good. Corrosion resistance is excellent and elevated temperature properties do not fall away rapidly. This class has not gained wide acceptance due, primarily, to poor weldability and modest strength. The well known

TABLE 3
CHEMICAL COMPOSITION OF SOME HOT WORK TOOL STEELS

Designation	AISI Type	Typical Chemical Composition %							
		C	Mn	Si	Ni	Cr	Mo	V	Other
Peerless 56		0.40	0.55	1.00	—	3.25	2.80	0.35	
Potomac M	H-13	0.40	0.35	1.00	—	5.25	1.25	1.00	
*Crovan	H-13	0.34	0.40	1.07	—	5.08	1.39	0.92	
Vasco Jet 1000	H-11	0.40	0.30	0.90	—	5.00	1.30	0.45	
*Avrocan M7-12	H-11	0.41	0.38	1.00	—	4.92	1.35	0.48	
Potomac A	H-11	0.40	0.30	0.90	—	5.00	1.30	0.50	
Thermold J		0.50	0.35	1.10	1.40	5.00	1.30	1.00	
Halcomb 218	H-11	0.40	0.40	1.05	—	5.00	1.35	0.50	
Halmo	H-15	0.35	0.28	0.72	—	5.16	5.14	0.66	
*HW-24		0.35	0.57	1.20	—	3.48	4.24	0.81	
*B 47		0.41	0.22	0.20	—	4.17	0.39	2.15	W 4.49 Co 4.26

*Actual Atlas Heat Analysis, balance from Reference 14.

A-286 alloy is considered to be a superalloy and is excluded from this discussion, although it is basically a highly alloyed PH austenitic stainless steel.

Hot work tool steels

The recent interest shown by the aircraft industry in hot work tool steels has been due to their ability to retain higher strength levels at elevated temperatures than either the constructional or the stainless steels. The 5% chromium hot work steels of the AISI Types H-11 and H-13 have to date dominated the picture, particularly the Type H-11 (see Table 3). This latter type appears to have the best combination of properties among the hot work steels to comply with present specifications in the 260,000-300,000 psi range.

Numerous articles[12, 13, 14] have been published listing the properties and advantages of hot work steels for aircraft applications. Some of these advantages and disadvantages are listed below in comparison with the stainless and the low alloy construction steels.

Advantages

(1) Highest ultimate tensile strength tested at 400/1,000°F (see Figure 2).
(2) Highest 1,000 hour stress rupture strength at 1,000°F.
(3) Air cooling reduces quenching stresses, thus practically eliminating distortion.
(4) Secondary hardening requires a higher tempering temperature which further reduces residual stresses.
(5) Multiple tempering removes practically all traces of retained austenite. This benefits ductility and fatigue resistance by avoiding the possibility of retained austenite transforming to martensite in service.
(6) Higher chromium content improves oxidation resistance.
(7) They are not generally considered to be notch sensitive.
(8) Welding and other fabricating procedures are well established.

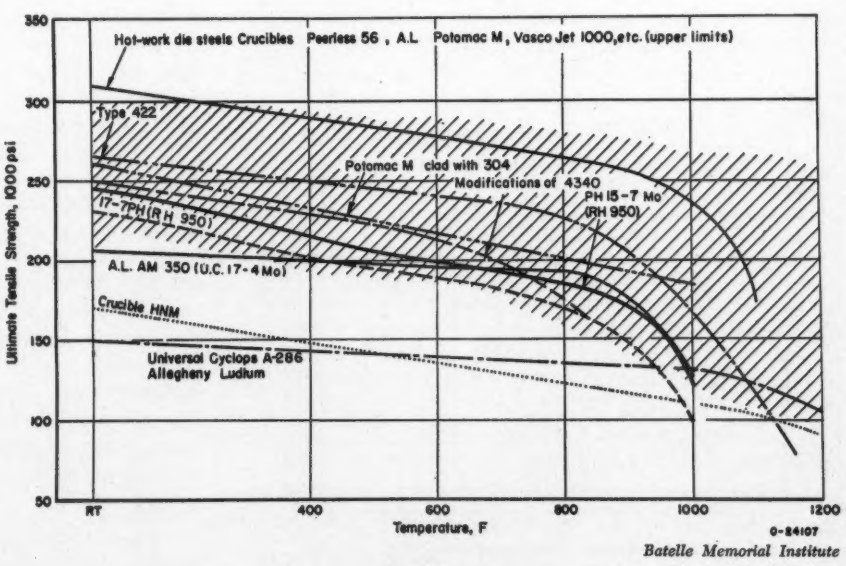

Figure 2
Comparison of typical tensile strengths of presently available steels with airframe needs[14]

(9) Room temperature impact and ductility properties at high strength levels are good.

(10) Fatigue test properties obtained to date are excellent.

(11) Higher silicon contents are not required; therefore, the cleanliness problem is benefited.

(12) Relatively low alloy content reduces cost and use of strategic alloy additions.

Disadvantages

(1) The major disadvantages of the hot work tool steels is their lack of corrosion resistance. This fact has restricted their use for external applications.

(2) Susceptibility towards decarburization, especially in thin sections, requires that considerable care be exercised for its control.

(3) It is generally assumed that these grades are prone to hydrogen embrittlement. Precautions are taken to ensure adequate baking after plating.

(4) The hardening temperature of 1,850°F tends to accelerate decarburization, compared with the lower hardening temperature of the constructional steels. Adequate high temperature hardening equipment is required.

A number of methods have been used to improve the corrosion resistance of hot work steels. Cladding with Stainless Type 304 (5% each side) has been done, but with an appreciable sacrifice of strength. Heat resistant enamels are claimed to be satisfactory even at the 800/900°F range. Ceramic coatings have also been recommended for 1,000/1,200°F service. Hot dip aluminizing, diffused nickel-cadmium plating, and vacuum depositing cadmium have all met with some success. While all of these methods improve corrosion resistance, they do involve added complications, expense, and loss in strength, particularly in thin sections.

Decarburization reduces fatigue resistance and strength. This effect increases as the section size decreases; therefore, in the case of thin gauges, the problem can become serious. The decarburization, which is necessarily present with mill bark on rolled or forged stock, will usually be removed by subsequent fabricating operations. On the other hand, the decarburization which might be produced by the aircraft concern's heat treatment must be minimized by careful treating in neutral salt baths, endothermic atmospheres, or by the use of a protective coating on the surface.

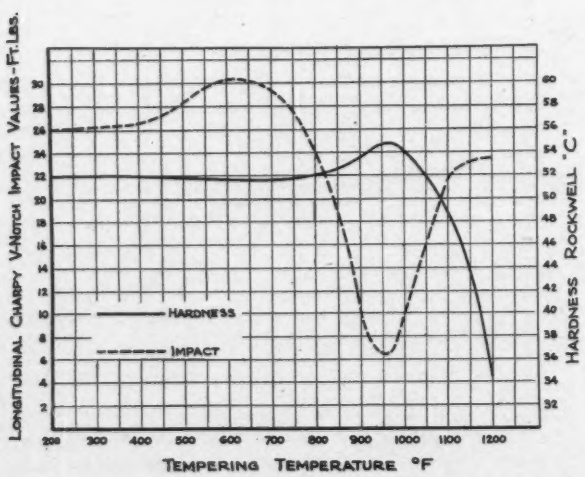

Figure 3
Hardness and impact properties of Crovan after double aging at various tempering temperatures

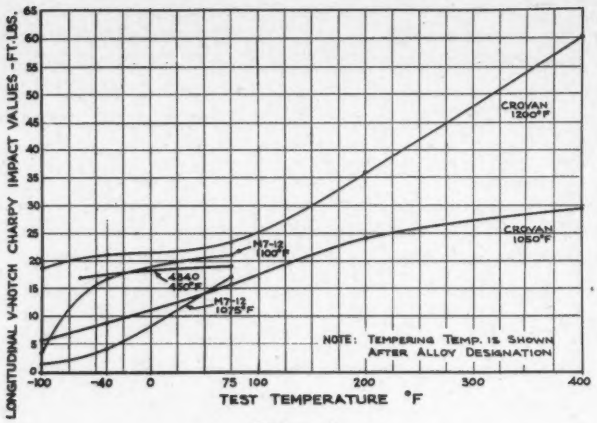

Figure 4
Impact properties at various testing temperatures

Available information is not too abundant on the impact properties of hot work tool steels, especially at low temperatures. While much is made of the 500°F embrittlement zone of the low alloy constructional steels by the hot work advocates, the existence of a low impact zone with hot work steels is usually ignored. Figure 3 illustrates this drop of impact resistance corresponding to the region of maximum hardness obtained at the secondary hardness peak[15]. This impact valley, however, does not present a problem, as normal hot work practice always requires tempering to hardness values beyond the secondary hardening peak, not only for toughness but also for better structural stability.

Low temperature impact properties are of interest to designers for items like landing gear components which may have to operate from northern fields. Results from preliminary tests shown in Figure 4[15] have been responsible for a comprehensive program on this phase by the Department of Mines and Atlas Steels. Indications point to the fact that 4340 is superior to the H-11 and H-13 hot work tool steels when tested at low temperatures at approximately the same hardness levels.

PROBLEMS OF THE STEEL MANUFACTURER

A number of excellent publications have discussed the aircraft industry's processing problems and performance requirements for present and future ultra high strength steels. The steel producer also has his share of difficulties. A few of these problems which are becoming more acute as strength requirements increase will be touched upon briefly.

Cleanliness

The requirement for a high standard of cleanliness in aircraft steels, particularly for those components to be heat treated to high strength levels, is an absolute necessity. The melting practices to achieve this quality and

the inspection methods to establish the required standard present a number of difficult problems. Dispensing with the melting problems for the moment, the matter of specifications and inspection methods will be considered. The term "aircraft quality" is more or less meaningless. About all it means is that the producer followed certain practices and took certain precautions, and the consumer has paid extra for additional inspection and care.

The conventional tests are all right as far as they go, but they are limited. The microscope can determine the existence of and identify inclusions, but only views an extremely small area. The magnetic-particle stepdown test covers a much larger area than the microscope, but it, too, is limited by practical considerations. It is, however, an extremely sensitive test on a well-machined surface. The fact remains that, after all the customer's tests are complied with at the mill and the steel maker has, from experience and statistical studies, selected the ingots and portions thereof for use and test, the actual material shipped has not been checked internally. The use of the contact supersonic tester on a mill surface is not sensitive enough to locate the size of the inclusions we are concerned with. This situation causes friction when a customer finds inclusions in a finished part or a failed component. He quite naturally feels let down because the steel was not what he considered to be aircraft quality. On the other hand, the steelmaker feels he conscientiously tested the material and that the customer, if he looks long and hard enough, is bound to find an inclusion. On occasion, steel mill metallurgists may feel that the role of inclusions has been over-played. It is perhaps unfortunate that residual stresses do not betray their presence in such a concrete manner as do inclusions. When the cause of failure is obscure, it is much more satisfying to have a culprit that can be photographed. Superimposed on this situation is the immersion ultrasonic inspection standard. Here, we have a tool that views volume as opposed to area, and is now sensitive enough to locate extremely small inclusions. The steelmaker has possibly two immediate alternatives: preselect to the best of his ability by use of the conventional tests, or double melt. A third alternative might be for the steelmaker to ship immersion ultrasonic tested stock. The setting up of an immersion ultrasonic standard that is acceptable to both producer and consumer and clearly defines cleanliness is difficult.

An interesting approach from the steelmaker's point of view is being taken by Avro Aircraft Limited in their experimental immersion ultrasonic tests of full size bars, rounds, and flats. Briefly, this test is an attempt to determine what is the least expensive metal surface the mill can ship steel with that permits sensitive enough immersion ultrasonic readings to detect inclusions of a size that might be cause for later rejections. It is intended that a bar be examined full length and the poorest areas marked. The poorest section would then be cut out and examined by microscopic and magnetic particle testing. This test would determine whether the quality was aircraft, standard, or scrap. If the quality was not aircraft, then further cutting and testing would be carried out at the next poorest location. This type of test, if proven feasible, could eventually be done at the mill, thus enabling the steel producer to ship only immersion ultrasonic proven quality stock to the aircraft customer.

Vacuum melting

Increasingly severe cleanliness specifications and the introduction of immersion ultrasonic inspection standards may require a double melting procedure to obtain a more uniform quality from heat to heat, and avoid excessive inspection and pre-selection of single air melted arc heats. A typical double melting practice would involve the re-melting of air arc melted electrodes under vacuum in a cold mould consumable electrode furnace.

Tramp elements

It is suggested that not enough attention has been given to establishing the effect of tramp elements upon mechanical properties, such as toughness and transverse ductility. This could conceivably force the steelmaker to avoid scrap in favour of a virgin melting charge.

Transverse bend tests

The standard tensile test for steels about 55/56 Rockwell "C" becomes more and more meaningless as hardness increases. It is recommended that a type of bend test be adopted in its stead[16].

Ductility specifications

Improved elevated temperature properties at high strength levels invariably necessitate a higher alloy content and lowered ductility. Some re-evaluation of ductility requirements and methods of testing eventually will have to be undertaken. The aircraft industry has been a leader in this matter of utilizing structural parts of low elongation levels. For this reason, if for no other, the critics of high ductility requirements should be willing to wait until an adequate amount of testing correlated with service experience has been acquired. The main arguments advanced in favour of lowered elongation values are that lower values are already specified for high strength aluminum and that most components cannot permit more than about 3% elongation without malfunctioning. Nevertheless, aside from fabrication, some ductility is helpful for the re-distribution of stress to accommodate load adjustment and give a rough idea of the reserve of plasticity available for insurance against shattering fracture in an accident.

Notched bar impact testing

This test is normally used to give an indication of toughness or, more specifically, a check on metal quality, quality of heat treatment, and relative sensitivity to clearage under conditions of suddenly applied stresses[16]. The notched bar testing at very high hardness fails to give any significant results and the un-notched bar or round lacks sensitivity and results are erratic. Other tests sometimes used for an indication of toughness are the static torsion, torsional impact, and the transverse bend test. None of these tests are fully acceptable to all concerned. The problem is not a matter of defining the vague term "toughness", but of determining what properties need evaluating and what is the best test method to achieve this purpose.

FUTURE TRENDS

One aspect that may influence the future choice of an alloy is the anticipated length of life of the vehicle

involved. So far, our alloy steels have been discussed with a high performance manned aircraft or a recoverable missile component in mind. Here, the use of expensive alloys capable of retaining their properties over a period of hundreds of hours is justified. The one shot missile, whose life may be only a matter of minutes, presents a different picture. In this case, it is desirable to use the most non-strategic and non-critical of materials. It is possible that many of our existing alloys may be up-graded for this type of application.

Although the constructional steels are capable of achieving over 300,000 psi tensile with very low alloy content, their lack of corrosion resistance and inability to retain their strength even at moderate temperatures limits their potential for further development. Where good ductility and low temperature impact resistance are of prime importance, certain of these steels, such as EN30B, may warrant further study.

The work hardening austenitic stainless steels typified by Type 301 are not capable of attaining strengths much over 260,000 psi and, as a result, are restricted to those lower strength applications where they possess a combination of many desirable properties.

Currently, the 12-13% chromium martensitic grades of stainless are receiving attention. Hardenable Types 410 and 420 are capable of being alloyed with Mo, V, W etc for improved elevated temperature properties. While the use of a hardenable stainless steel is attractive, two problems are encountered: firstly, lowered corrosion resistance due to the increase in alloy content and, secondly, the ductility and toughness results are relatively low with 12-13% chromium plus sufficient carbon for very high strengths.

The relatively new precipitation hardening grades of stainless steel are capable of extensive development and most producers of stainless are following this field closely[18]. Other than 17-4 PH, the martensitic group, due to poor formability, are not receiving much attention. Somewhat the same situation exists with respect to the austenitic types. While the 18-8 austenitic type of alloy is very desirable from the workability aspect, the stability of its austenite may be easily unbalanced by alloy additions capable of producing strengths in excess of 200,000 psi. As for all types of PH stainless steels, a great number of possible precipitation hardening additions remain to be investigated. Effective additions, with respect to hardness, tend to lower the corrosion resistance and unbalance the structure to such an extent that excessive amounts of delta ferrite or martensite may be formed. Precipitating elements currently added to commercial steels are capable of increasing the hardness from 10 to 15 points Rockwell "C". However, experimental steels have exhibited an increase in hardness due to precipitation as high as 30 points Rockwell "C". Unless small amounts of a new addition prove much more potent, it is difficult to visualize how the solution annealed austenite may be sufficiently hardened by precipitation alone to produce high strengths. It is possible to obtain hardnesses as high as 58 Rockwell "C" from a low carbon 18-8 base, but the relatively high "as-quenched" solution annealed hardness prior to aging leaves something to be desired regarding ductility.

To a large extent, the semi-austenitic steels overcome the disadvantages of the other PH types by having a soft and readily-workable austenitic structure when annealed, and which can be subsequently hardened by a martensitic transformation. Additional hardness from precipitation may then be superimposed on the hardness of the tempered martensite during aging. This supplementary precipitation hardening does not, therefore, require the amount of alloy to achieve over 40 Rockwell "C" as in the case of the austenitic grades because precipitation is starting from an initially harder matrix.

New combinations of precipitating additions are being investigated in the hope that higher strengths and improved elevated temperature properties may result. Quite apart from these major aims are several other factors to be considered in arriving at a new precipitant. Ideal additions should be non-strategic, low-priced and not affect the corrosion resistance adversely. Certain elements, such as copper, are not welcome in a melt shop due to the chances of scrap contamination. Good weldability restricts the choice of alloy additions. Nitrogen and phosphorus, for example, tend to complicate welding procedures, and elements such as aluminum and titanium are subject to oxidization during welding. It is difficult to control the amount and distribution of such elements as aluminum, which presents a quality control problem in obtaining uniform response to heat treatment. The present high strength AM-355 and 15-7 Mo type alloys are still capable of further improvement in strength.

The basic 5% chromium hot work tool steels are still undergoing development and modification. One important contribution of this work has been the realization that we still require more basic information on the heat treating of these steels and its effect upon structure. One case in point is the problem of meeting transverse ductility specifications. Rather complicated conditioning heat treating cycles have often been found necessary to control the precipitation of carbides at the grain boundary in order to improve the transverse properties.

Effort is being expended towards improving upon the elevated temperature properties of the 5% chromium hot work steels beyond 1,000°F (Figure 5). The older high tungsten hot work steels are not generally favoured due to their lack of toughness and duc-

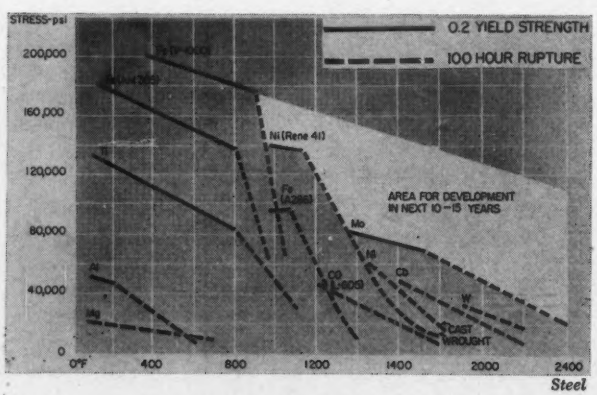

Figure 5
Where metals are today[20]
(best alloy of each system)

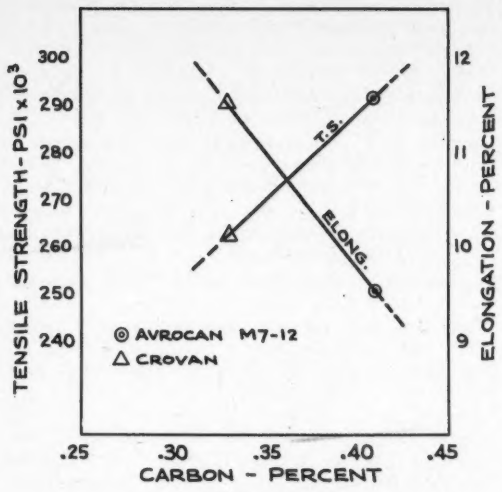

Figure 6
Tensile strength and elongation as a function of carbon content. Both alloys air quenched from 1,850°F, and double tempered at 1,050°F

tility, aside from increased weight and cost. One approach has been towards the more highly alloyed and complex compositions containing some tungsten along with chromium, cobalt, and vanadium, such as in the B-47 type of super hot work steel. This alloy at 1,100°F shows a tensile strength of 210,000 psi[14].

A second approach has been towards increasing the molybdenum content of the chromium hot work steels up to the point where molybdenum becomes the primary alloy addition. Peerless 56, HW-24, and Halmo would represent this line of attack. Working from the other direction is the investigation of low carbon molybdenum high speed steels, or semi-high speed steels. HTB-2, HTB-3, and AISI Type H-41 (Mohican 6) would be representative of the semi-high speeds.

Progress in this super hot work semi-high speed area is largely dependent upon a better understanding of the function and interactions of elements such as chromium, tungsten, molybdenum, vanadium, and cobalt, plus carbon content on strength and retardation of softening.

The optimum carbon content for these high strength grades compatible with adequate strength and reasonable ductility has not been completely established[19]. A. J. Lena has indicated that for a Potomac hot work tool steel the elongation and strength curves cross over at 0.40%C. Limited Atlas data on Crovan and Avrocan M7-12 would indicate that the cross-over is between 0.35% and 0.40% C (Figure 6). This data indicates that the strength obtainable would be limited by the ductility requirements. Figure 7 illustrates this situation in a slightly different manner. The fact remains that high strength and high ductility are not fully compatible.

The AISI Type M-2 molybdenum high speed steel possesses the highest strength of any of our commercially available materials. This, the most common of all present-day high speed steels on the North American continent, is capable of attaining an ultimate bend strength of 650,000 psi[16]. This strength is obtained not at maximum hardness, but by tempering to a small restricted temperature range just over the secondary hardness peak. This phenomenon may also be observed with hot work steels. Tempering below or above this optimum tempering range results in a drastic loss of strength. The use of an M-2 high speed tool steel is not recommended for general aircraft applications due to its low ductility, which would necessitate a complete re-evaluation of present design criteria. This remarkable steel might, however, point the way towards a lower alloyed family of steels adaptable to ultra high strength aircraft components.

CONCLUSIONS

(1) The most promising ultra high strength steels for the immediate future at moderately elevated temperatures appear to be the precipitation hardening, semi-austenitic stainless steels and the super hot work, semi-high speed tool steels. The semi-austenitic stainless types have a distinct advantage in thin sections for external applications: on the other hand, the super hot work semi-high speed are unequalled for load carrying ability at temperature and are most suited to thicker sections where some protection against corrosion may be afforded.

(2) The future acceptance and development of these grades will require some re-evaluation of present ductility specifications, testing methods, and melting practices.

(3) The future development of ultra high strength steels will depend largely on our obtaining a better understanding of the role being played by individual alloying additions and their interations with regard to strength and retardation of softening.

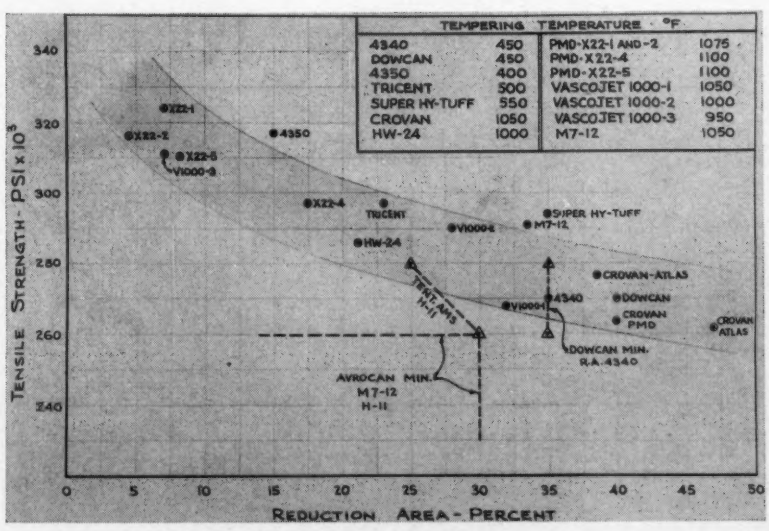

Figure 7
Ultimate tensile strength of some high strength steels versus reduction of area (longitudinal properties)

ACKNOWLEDGMENTS

The author wishes to express his thanks to Mr. E. G. Schempp, Atlas Steels Limited, and Dr. W. A. Morgan and his associates at the Physical Metallurgy Division, Department of Mines and Technical Surveys, Ottawa, for data used in this paper. Permission to mention the immersion ultrasonic tests being conducted at Avro Aircraft Limited was kindly granted by Mr. R. Smallman-Tew of that concern.

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PILOTING RESEARCH AIRCRAFT†

by J. Walker*

National Advisory Committee for Aeronautics

INTRODUCTION

HISTORICALLY, the pilot has been at the forefront of new developments in aircraft and flight testing, aiding in the design and suggesting improvements. Last February at the Mid-season Meeting, Mr. A. W. Blackburn of North American Aviation gave an excellent presentation[1] of why this concept of the man participating as an integral part can profitably be continued for more advanced vehicles. Nevertheless, some proposals virtually eliminate the man, except as a passenger, in hypersonic and orbital vehicles. These proponents of automaticity apparently believe that the problems and tasks of controlling the vehicle are becoming so difficult that the pilot cannot cope with them. This belief is far from new; for instance, it was advanced before supersonic flight was achieved as a reason why Mach 1 could never be surpassed. Actually, we, the pilots, thought we were doing a fairly creditable job and one that should have guaranteed us first consideration for the next advanced task. Instead, we find ourselves forced to justify our very existence.

I will cite, in this paper, some examples of control problems from our experience in flying advanced research and production aircraft at the NACA High Speed Flight Station. These problems range from those which the pilot could cope with himself to those which occurred without warning and, once encountered, were practically uncontrollable. These problems were not solved by eliminating any piloting function or responsibility. Instead, each of the problems was solved by improved design, better airborne and ground aids, and improved methods of flight testing, retaining the pilot as an integral part of the control system. In addition, some problems will be described which are foreseen for the future and should be corrected in the design stage.

RESEARCH AIRCRAFT

Our primary responsibility at the High Speed Flight Station is operation of special research aircraft which are designed and constructed to obtain flight data of advanced nature. These vehicles are built for as high performance as is practicable considering the state of knowledge, tools, and materials (Figure 1). Each aircraft has been conceived with specific objectives in mind. The X-1 had the objective of obtaining supersonic flight.

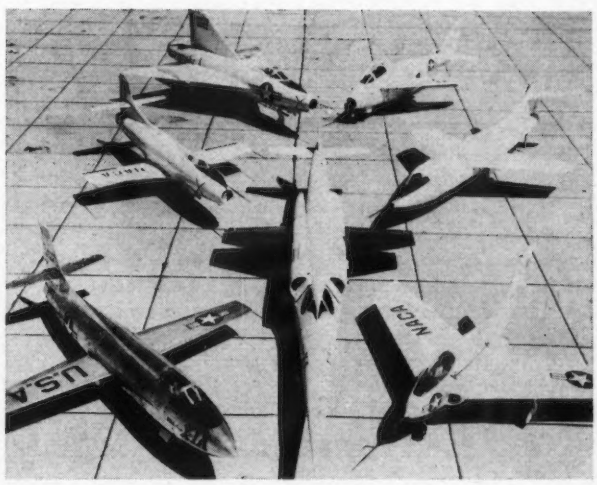

Figure 1
Research aircraft

The D-558-II was used to study the application of wing sweepback to problems of transonic and supersonic flight. The X-5 had the capability of variable wing sweep. The X-3 was a return to straight wings for sustained supersonic flight, but they were very thin and stubby. The X-2, not shown, was the first aircraft with performance capability which required consideration of aerodynamic heating in selection of materials. The X-1 series, D-558-II, and X-2 were air-launched rocket-powered aircraft. The others were jet-powered and took off from the ground. Following tests using this group of research aircraft, one of the original X-1's was modified for increased propellants and incorporated a thin wing, thus raising its performance capability, and was redesignated the X-1E. Now in process of construction is the X-15 which will once again increase the speed and altitude performance and actually come to grips with the heating problem. Note that each of these aircraft is flown and controlled by a pilot. Many do not even have power-control systems, and only one has dampers.

DEVELOPMENT OF TECHNIQUE

Research flight testing has always been supported by every available tool. Large advances have been made since the day when the pilot wrote all the data on a knee pad during the course of a test flight. Then, the pilot was limited to the use of wind tunnel information

†Paper read at the Joint IAS/CAI Meeting in Ottawa on the 8th October, 1958.

*Chief Test Pilot, High Speed Flight Station.

and his own previous experience as guides to what to expect of a new aircraft. Internal recording of measurements, use of radio for verbal transmission, and telemetering have relieved the pilot's data-recording load during flight. This is a necessity, since the portion of flight available for obtaining high speed flight information is often shorter than the time devoted to one data point in the early days. Formerly, it was desirable to fly the airplane with speed and altitude constant during a particular test run, attempting to change only the variable of interest. Now, the techniques for obtaining data "on the run" have been developed so that the pilot need only have the airplane steady at the selected load factor, prior to the initiation of the required maneuver. Minor variations of speed or altitude, or both, are accepted in the interest of more rapid accomplishment of test maneuvers for stability and control testing. Several standard maneuvers are used to determine static and dynamic stability characteristics as well as handling qualities and performance. For instance the wind-up turn; the lateral, directional, and longitudinal pulse; aileron roll; sideslip; rolling pullout; push-pull or roller coaster are completely adequate, with variations, for obtaining an accurate picture of the airplane derivatives and characteristics if the aircraft is stable over its entire usable flight envelope.

Step by step approach

With reference to aircraft control problems, one of the earliest encountered as the performance of aircraft increased was the decrease of longitudinal control effectiveness approaching the transonic speed and simultaneous apparent increase in longitudinal stability or "tuck". An illustration of some variations of this characteristic is shown in Figure 2 in which stick force is plotted against Mach number. This figure shows three different types of longitudinal trim curves obtained for aircraft capable of attaining supersonic speed. The dashed curve represents an earlier control system version where there was a strong tuck followed immediately by an abrupt nose-up tendency, and then a gradual increasing longitudinal stability trend. This is illustrative of an elevator control.

The other two curves represent aircraft with all-movable stabilizer control; however, for the one represented by the solid curve the magnitude is reduced

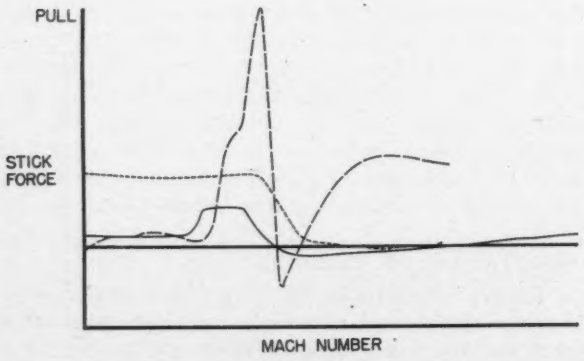

Figure 2
Longitudinal trim variations with Mach number

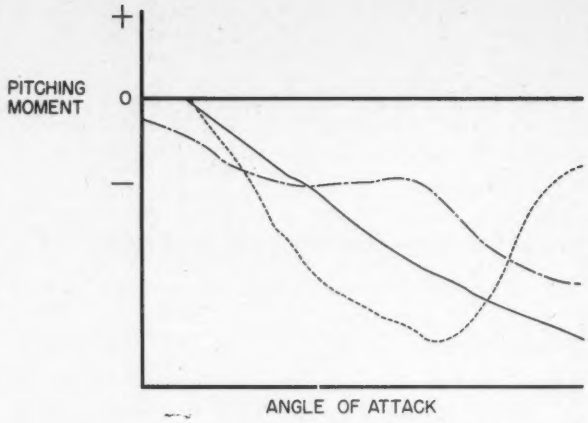

Figure 3
Longitudinal stability characteristics

markedly, and for the dotted curve, illustrating an improved control system, the tuck is not even noticeable. The pilot's problem in obtaining useful data in the transonic region was the inability to obtain a steady trim point. However, testing could be done in this area because the pilot could ease up on the particular point and feel-out the aircraft and could decelerate if difficulty were encountered. This made possible the use of a standard testing procedure whereby gradual logical steps of progression are used. If an area of particularly large change or difficulty is encountered, the steps are shortened to enable work-up of the information on the ground and a prediction of probable trend before going ahead in flight. This procedure is easily accomplished because it is a static stability investigation where the aircraft is intentionally held steady at 1 g.

Figure 3 shows three pitching moment curves with pitching moment plotted against angle of attack; the dotted line represents an obviously unstable pitching moment curve. The dashed curve, although never completely unstable, still illustrates a reduction of stability, and the solid curve represents a stable airplane. The problem involved here, of course, is that in maneuvering flight, as the angle of attack is increased, the aircraft with unstable variations remains stable until it rapidly pitches up and the pilot is forced to reverse the stick in attempting to keep it from getting out of control. If the technique of gradual stepwise test progression were applied to this type of problem, the change could be approached more or less safely and, at the same time, information could be obtained which would lead to the belief that the next step would be less safe than the one just accomplished. This was actually the case and, in attempting to reduce the dynamics of the maneuver to a minimum, very slow rate of control application was used. However, when decreased stability was indicated by the curve, the airplane would still tend to pitch up and it was problematical from one test to the next whether the rapid application of corrective control would be successful in time. We see that when rapid dynamic motion takes place, the pilot begins to be limited in his ability to evaluate and control.

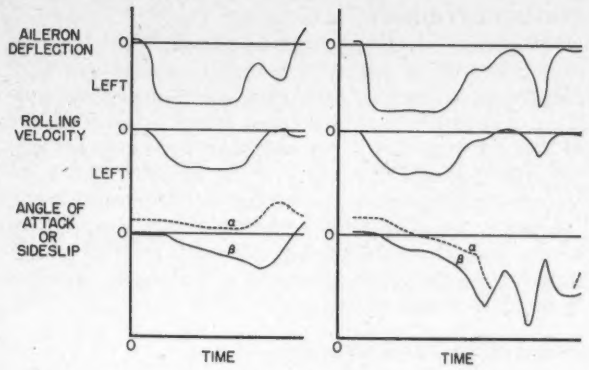

Figure 4
Rolling characteristics

Figure 4 shows a problem which includes even more dynamic characteristics and actually might be considered to exceed the pilot's control capabilities. This is an illustration of the inertia coupling problem. Two examples of rolls at essentially the same conditions are shown. Plotted are the aileron input, roll rate, angle of attack, and angle of sideslip. The only apparent initial difference is the small increase of aileron input and the resulting increased roll velocity. Actually the roll on the right may have felt better to the pilot than that on the left; better, that is, until recovery was initiated. The differences after that are obvious.

Ground Simulation

Since the pilot's capabilities are apparently exceeded, some additional method must be found to aid the pilot in determining how far he can go. We now come upon one of the modern giant steps in the evolution of flight testing. A three-way attack upon flight tests demonstrating or approaching this type of problem was evolved. A ground-based simulator using a computer is set up incorporating the characteristics of the aircraft as determined from wind tunnel model tests. The problem is set in and actually "flown" in order to obtain a rough idea of the particular problem areas where coupling could occur. During exploratory flight tests, these areas are avoided until flight-obtained derivatives are put into the ground simulator and flown. Then the flight tests proceed in the wake of the ground-simulator explorations, and the data obtained during the flight test are plotted for comparison with those obtained from the ground simulator (Figure 5). As the tests progress, the correlation of flight information with that on the computer is closely checked. In most instances as the test data follow and indicate the trend already determined from the ground simulation, the tests are terminated when it is apparent that either the stability or load boundaries are being approached. This system has proved to be extremely accurate. In contrast to the illustration in Figure 4 of tests which were made before the use of this technique, a roll demonstration utilizing the described technique was accomplished without the incidence of a single dangerous maneuver.

Such test technique is also of value in the region of directional stability at high speeds. It is possible to attain speeds such that the directional stability of the aircraft is so low that the initiation of any maneuver will result in complete loss of control of the aircraft. Figure 6 from Reference 2 illustrates this point. The directional stability parameter C_{r_β} at trim angle of attack and the directional period are plotted against Mach number for three altitudes. At the right is the speed limitation where directional stability is zero for trim angle of attack. The pitfall to be noted here is that any maneuvering requires increased angle of attack. This is represented by the dotted line labelled α_2. Hence, the maneuvering maximum speed is less than that for steady flight.

Observe also that the period lengthens both with speed and altitude. The pilot is led astray by the illusion of increasing stability, since his lateral acceleration clues are gone. This could happen also if he were diving to gain speed; for instance, it is possible to proceed along a constant period line in descending from 70,000 ft to 40,000 ft and approach dangerous conditions. Use of the same test technique for roll testing will aid in determining where these hazardous areas occur and assist in avoiding them.

Each of these foregoing problems is representative of those we have encountered while still utilizing the pilot.

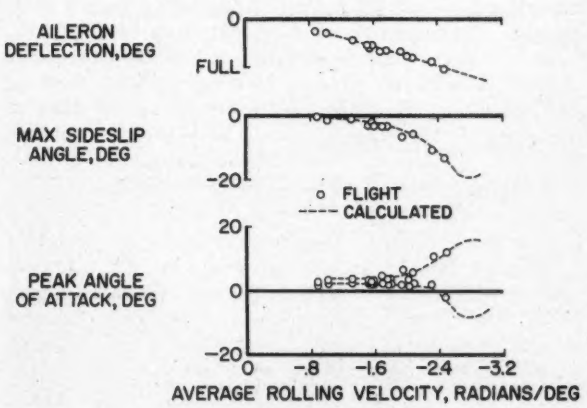

Figure 5
Flight-determined and calculated excursions in 360° rolls

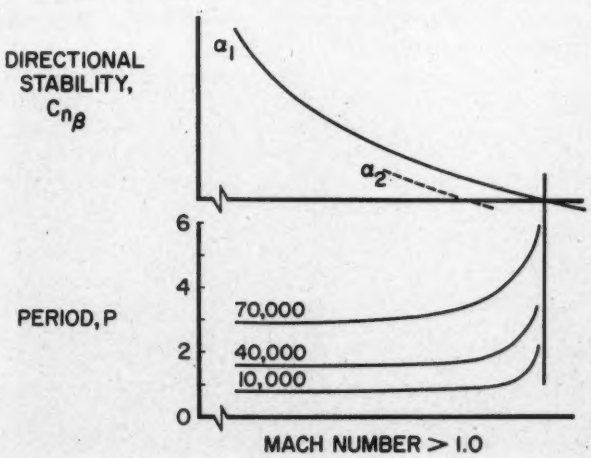

Figure 6
Directional stability and period as functions of Mach number

LANDING

One of the primary problems of the future will be that of landing. Though this problem is by no means new, it will become increasingly important as vehicles gain in performance. As mentioned earlier, hypersonic glide and orbital vehicles are being studied and planned for purposes of carrying man at higher speeds and altitudes, and into space. We find that emphasis is being placed so strongly upon maximum performance that all the powerplant and fuel are devoted to achieving the goal of altitude or velocity, and the vehicle is recovered without power, if possible. Experience has indicated that the problems of designing to greater performance goals and of combatting aerodynamic heating result in aircraft or vehicles of lower and lower lift-drag ratios. As the lift-drag ratio decreases, returning the aircraft to the ground safely becomes more than an exacting task. It becomes a problem requiring intense concentration, and small latitude of action is available even at relatively high altitude.

If we compare an approach at an L/D of 3 with that at an L/D of 6, it will be noted that from a given altitude, we can go only half as far in half the time at the same forward velocity. When consideration is given to actual forward velocities required, some chilling facts materialize. While the L/D = 6 aircraft allows a comfortable 4 minutes or so for descent from 15,000 ft, the L/D = 3 aircraft can descend in as little as 90 sec at a vertical velocity of 170 fps. Vertical velocity must be reduced to less than 9 fps in the last 15 sec. Because of lift characteristics the L/D = 6 aircraft will probably glide near maximum L/D, whereas the other will glide at still lower L/D than its maximum capabilities. Higher speed for the low L/D aircraft is a safety factor for dive recovery in order to be tangent to the runway prior to attaining minimum safe speed and to allow for increased drag, such as gear extension. However, this is predicated upon a runway length adequate to remove the mental strain involved in attempting to hit a runway with sufficient length remaining after contact to allow deceleration to a stop. In the final analysis this means, then, that a large area is preferable to a fixed runway for manual visual landing. Assistance by guidance equipment would only partially alleviate this problem, because, of course, the pilot is expected to accomplish the job if the guidance equipment fails.

It would appear that the common-sense solution to this problem would be to provide thrust for landings. Of course, the argument of weight is immediately raised. However, to the pilot, and perhaps to overall safety of operation, this engine might be considered indispensable equipment, even as the auxiliary power units or the control system.

HUMAN CENTRIFUGE

With the advent of large longitudinal accelerations caused by use of high-thrust rocket engines and high drag upon reentering the denser atmosphere, the problem of whether the pilot could retain control became of greatest importance. An additional ground-based tool for exploration of this problem was utilized. This is the human centrifuge at the Naval Air Development Center, Johnsville, Pa. The pilot is included in a closed loop with a computer and the centrifuge so that he is subjected to accelerations as a result of his control motions in response to instrument presentation stimuli.

INSTRUMENT PRESENTATION

Improvements in pilot instrument presentation have more or less kept pace with improvements in control systems, equipment, and ground aids. However, as vehicles go higher and faster, more and more dependence on the instrument presentation is required to accomplish the desired flight plan. In some instances, changes of attitude or angle of attack on the order of a degree or less must be detected and corrected. At present, more information is supplied to missile guidance systems than to pilots. If the same information were supplied to the pilot, the result would be a superior system with the pilot as an integral part of the control loop because of his reasoning capability and flexibility.

CONCLUSION

I have attempted to show that, although the human pilot has his limitations, experience has revealed that each time a problem has arisen, improvement of design, additional airborne and ground aids, and greatly improved flight test procedure have resulted in solution of the problem. These methods of solution attach no more stigma to the use of the pilot than to the use of pressurized cockpits and pressure suits. The process of flight testing using the method of progression by logical steps is still valid, with the length of the step dictated by the amount of knowledge or experience and the existence of any known or suspected problems. Recalling the immense increases in speed and altitude over the years with a man at the controls, it is concluded that piloted flight testing should be capable of assuring achievement of desired goals in the future.

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(1) Blackburn, A. W. – *Flight Testing in the Space Age*, Canadian Aeronautical Journal, Vol. 4, No. 4, p 126, April, 1958.

(2) Day, R. E. – *Flight Testing in Critical Stability Areas*, Quarterly Review Soc. of Experimental Test Pilots, Vol. 1, No. 1, Summer 1957.

PERFORMANCE OF A PERFORATED WALL INTAKE AT A MACH NUMBER OF 2.5†

by D. Brown*

Avro Aircraft Limited

SUMMARY

An axisymmetric internal compression intake was tested at a free stream Mach number of 2.5. The results show that starting of this intake is possible with a relatively good overall total pressure recovery being obtained but at the expense of a high bleed flow through wall perforations which are essential to starting. The total pressure recovery is sensitive to mass flow; a peak of 81% was measured with an exit opening 13% larger than the throat area and at this point a Mach number of about 0.25 occurs in the parallel part of the duct.

Using a theoretical approach the mass flow spilled through the wall perforations was estimated to be 23% of the intake mass flow. The penalty due to the bleed flow through the walls is of course shown up as a lowered net thrust of a ramjet or turbojet unit.

In addition, the bleed mass flow will be responsible for considerable additional external drag since a curved shock can be seen (between the lip and the first row of perforations) which will be strong near the wall.

For these reasons it appears that the intake in the form tested is not particularly well suited to the requirements of high speed flight, but by shrouding of the intake so that the bleed flow leaves by way of an engine ejector, it may be possible to eliminate some of the additional external drag referred to above.

INTRODUCTION

With the continually improving performance demanded of military aircraft, it became apparent some years ago that good intake design was as important as that of any other component of an air breathing propulsive unit. For flight Mach numbers up to 2 or so, a single cone spike inlet will give satisfactory total pressure recovery without too large a penalty in external cowl drag. Above $M = 2.0$ we are faced with the fact that good pressure recovery demands more than two shock waves for the deceleration to subsonic speeds and this calls for double spike geometry. This geometry unfortunately demands large external cowl angles with an attendant increase in external wave drag. For this reason there is now a good deal of emphasis on the design of wholly internal compression intakes (there is at least one aircraft flying with this type) since the external cowl can, with little difficulty, be made parallel to the flight direction.

In the test described, the intake model consisted of a de Laval nozzle designed for an exit Mach number of 2.5, reversed and joined to a divergent duct followed by a length of parallel pipe (Figures 1, 2 and 3). Mass flow

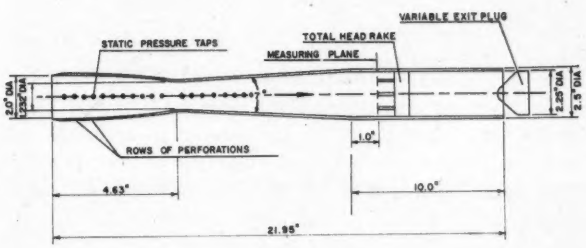


Figure 1
Schematic view of supersonic intake model

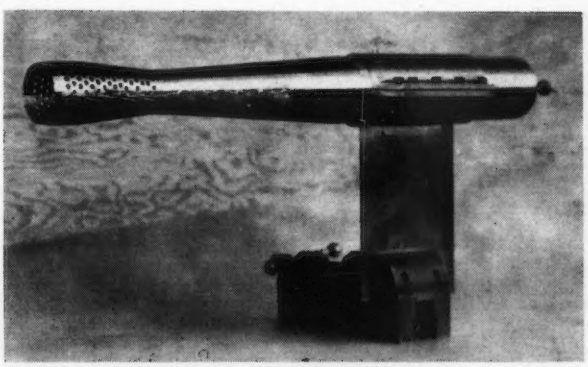


Figure 2
Front three-quarter view of model

Figure 3
Rear three-quarter view of model

†Received 2nd June, 1958. This paper is based on U.T.I.A. Technical Note No. 18[1].

*Project Research Group.

through the duct was controlled by a conical plug, movable in the axial direction.

No allowance was made for boundary layer growth and the intake lip area/throat area was 2.637, i.e. the isentropic expansion ratio of a nozzle designed for an exit Mach number of 2.5. Kantrowitz[2] shows that such an intake will not start owing to the formation of a normal shock ahead of the lip, unless the throat area is increased such that lip area/throat area $= 2.637 \times P_1/P_o$ where P_1/P_o is the total pressure ratio across a normal shock at a free stream Mach number of 2.5. For $M_o = 2.5$, $P_1/P_o = 0.499$ and the throat area will have to be approximately doubled for starting, i.e. for the normal shock ahead of the lip to move downstream of the throat. Opening up of the throat is not a good method of starting since Mach numbers down the inlet will be high and a rather strong normal shock will terminate the supersonic flow in the diverging part of the diffuser, unless the mass flow through the duct is kept small. Flow through the throat will be supersonic. In addition, the shock is only stable when downstream of the throat[2, 3] since when upstream of the throat any disturbance tending to reduce mass flow will force the shock towards the lip, thereby strengthening it with attendant increased total pressure loss and reduced mass flow, which enhances the expelling process.

A method of facilitating starting used by NACA[4] entails perforating the walls to bleed off the excess mass flow that the throat is unable to pass. An outflow results since for any shock position inside the intake there is a positive static pressure differential across the walls.

For a fixed amount of perforation the bleed mass flow will increase as the exit area is decreased, to match the new position of the shock which has moved upstream. Perforations downstream of the new shock position will now pass a greater mass flow than previously; thus mass flow equilibrium is maintained and the shock is stable in any position upstream of the throat. Figure 4 schematically represents the intake operation.

Following starting of this type of inlet the major sources of loss in total pressure from lip to throat are wall friction and the local oblique shock which occurs on the downstream side of each perforation. This shock loss may be minimized by using as small a perforation as possible.

LIST OF SYMBOLS

M	Mach number
S, A	area
P	total pressure
d	throat diameter
x	shock wave position
ξ	static tap position
Q_A	discharge coefficient of holes downstream of normal shock $= \dfrac{\text{Effective area of hole}}{\text{Geometric area of hole}}$

Subscripts

o	lip or free stream
1	throat of diffuser
E	exit
P	perforated

MODEL DESCRIPTION

In view of the possibility of aerodynamically blocking the 16 inch square tunnel section and the requirement for a short model, an intake lip diameter of 2.0 inches was taken and the converging part of the inlet was designed by Foelsch's method[5] as a nozzle with an expansion ratio and internal wall contour to give shockless flow at a Mach number of 2.5.

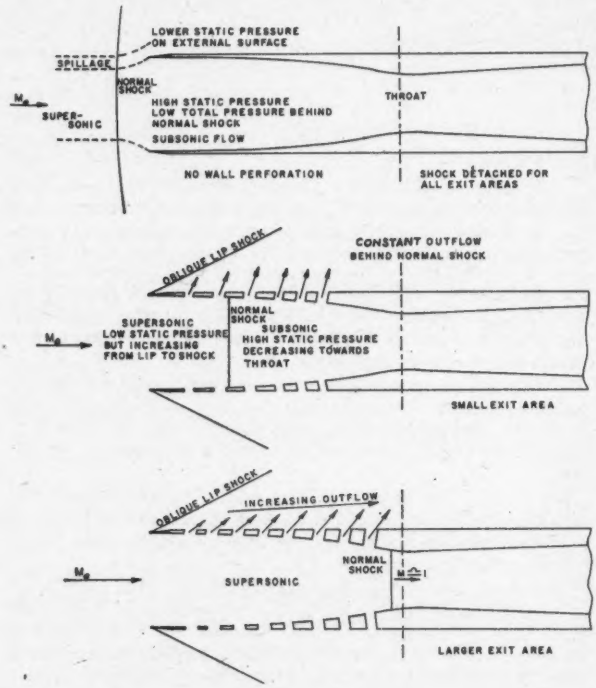


Figure 4
Schematic representation of operation of a perforated wall, internal-compression inlet

Figure 5
Sub-assembly of model showing total-pressure rake

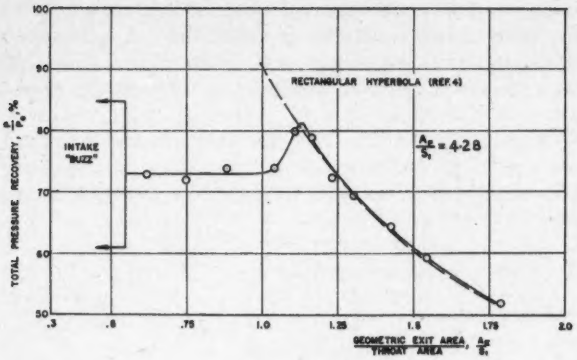


Figure 6
Total pressure recovery versus exit area

Following the throat, a conical diverging duct of 7° included angle was used, which was attached to a parallel section of 2.25 inches diameter. In Figure 1 the leading dimensions of the model are shown. Figures 2 and 3 show two views of the complete model and Figure 5, a sub-assembly.

A total of twenty-eight pressure tubes was installed. At the measuring plane (see Figure 1) five total-pressure tubes were mounted on a rake spanning the cross section of the parallel duct and four static taps were drilled in the duct wall in the same plane. The remaining nineteen tubes were connected to a line of static taps in the diffuser wall. Eleven of these taps were spaced ⅜ inches apart, upstream of the throat with the leading one 25/64 inches behind the lip, and eight were placed downstream of the throat ½ inch apart. No tap was placed at the throat inasmuch as the design of the converging part of the diffuser incorporates a steep pressure gradient near the throat and it was thought that a slight inaccuracy in placing a tap here would give misleading pressure measurement. All pressures were recorded on a multitube mercury manometer bank.

RESULTS

The variation of total pressure recovery with exit area is shown in Figure 6. An optimum total pressure recovery of about 81% was obtained with an exit to throat area ratio of 1.13, and the final perforation to throat area ratio of 4.28. At exit area ratios greater than 1.15, the variation of pressure recovery with exit opening is approximately represented by a rectangular hyperbola, as was found for area ratios greater than 1.2 reported in Reference 3.

Pressure recovery for area ratios less than 0.6 are not shown, owing to the emergence of unsteady flow in this region which grew more severe as the exit was closed. Complete expulsion of the normal shock, as can just be discerned in Figure 7(a), was obtained at exit ratios smaller than about 0.1.

Figure 7(b) shows unstable operation of the intake with the same amount of perforation – 9 rows out of 16 – as for Figure 7(a), but with an exit area ratio of 0.63. In this case the 'buzzing' was much less severe and no expulsion of the normal shock could be detected.

Figure 7(c) is shown for the purpose of comparing the shock system with and without perforation. The exit

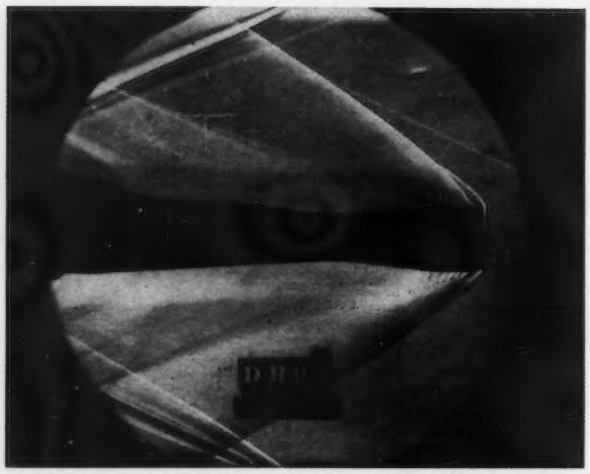
(a) Unstable shock system
$A_P/S_1 = 2.469$ $A_E/S_1 = 0.1$
$P_o = 73.5$ cm mercury

(b) Unstable shock system
$A_P/S_1 = 2.469$ $A_E/S_1 = 0.63$
$P_o = 73.5$ cm mercury

(c) Shock system with the unperforated intake
$A_E/S_1 = 1.787$ $P_o = 74.2$ cm mercury

Figure 7 (a, b, c)

(d) Shock system with some perforation
$A_P/S_1 = 1.425 \quad A_E/S_1 = 0.27$
$P_o = 74.2$ cm mercury



(e) Shock system with some perforation
$A_P/S_1 = 1.425 \quad A_E/S_1 = 0.63$
$P_o = 74.2$ cm mercury

(f) Shock system with some perforation
$A_P/S_1 = 1.425 \quad A_E/S_1 = 1.04$
$P_o = 74.2$ cm mercury

Figure 7 (d, e, f)

ratio was 1.787 and the inlet was not perforated. With the photographic plate used, the change in bow shock shape and position was almost undetectable when the area ratio was reduced gradually to zero (fully closed exit).

Figures 7(d), (e) and (f) illustrate the rearward movement of the shock as the exit area ratio was increased from 0.27 to 1.04, with a fixed perforated area ratio of 1.425.

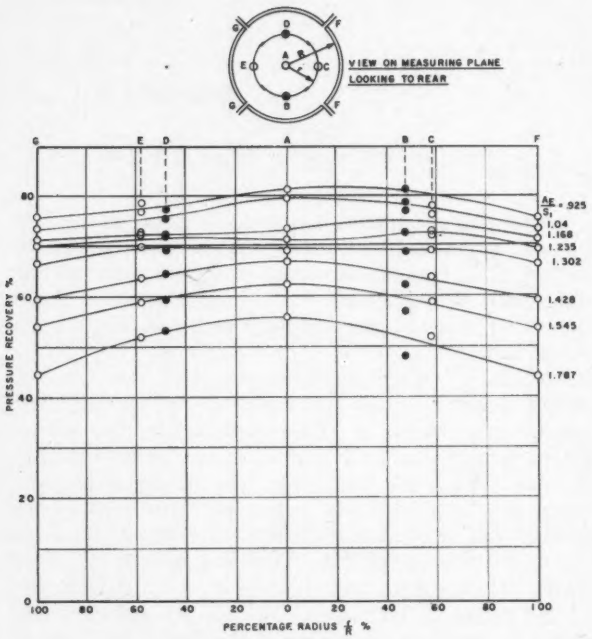


Figure 8
Pressure distribution across measuring plane

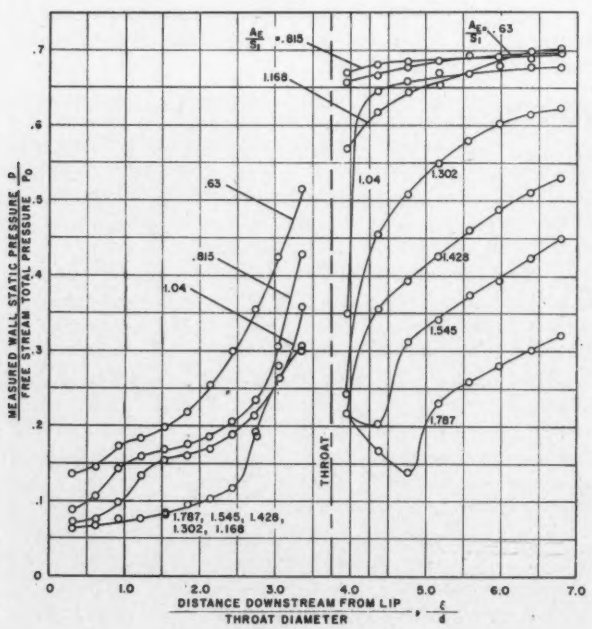


Figure 9
Static pressure distribution along intake
(final distribution of perforations $A_P/S_1 = 4.28$)

Figure 8 shows the variation of total and static pressure across the measuring plane. A certain amount of flow distortion is evident and probably arises from the non-symmetrical distribution of the perforations due to a strip of wall, about ½ inch in width, in which the static pressure taps had to be drilled and which was left unperforated on the right hand side looking rearwards (see Figure 2). For clarity, distributions at area ratios of 0.63, 0.764 and 0.815 are omitted from Figure 8.

Figures 9 and 10 show the variation of static pressure along the wall of the convergent-divergent diffuser. Figure 9 presents results for the diffuser in its final perforated state and gives an indication of the movement and position of the swallowed normal shock with varying exit area. At the low exit area ratios the shock is stationed between the intake lip and throat, evidenced by the steep rise in pressure upstream of the throat followed by a more gradual rise in pressure in the subsonic compression downstream of the throat. At these low exit area ratios the mass flow through the throat is so restricted that the normal shock is forced to stay upstream of the throat in order to increase the bleed through the subsonic portion of the perforation behind the shock and to establish a mass flow equilibrium running point. At the higher exit areas it is evident that the normal shock has passed through the throat since a drop in static pressure was found immediately downstream of the throat, followed by a sharp rise in pressure. As the duct diverges downstream of the throat a drop in static pressure is

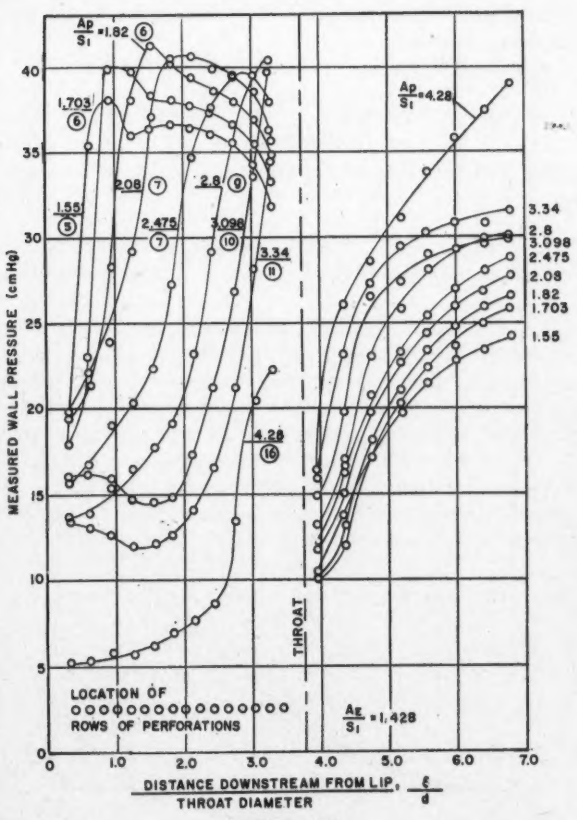


Figure 10
Static pressure distribution along intake indicating rearward movement of normal shock

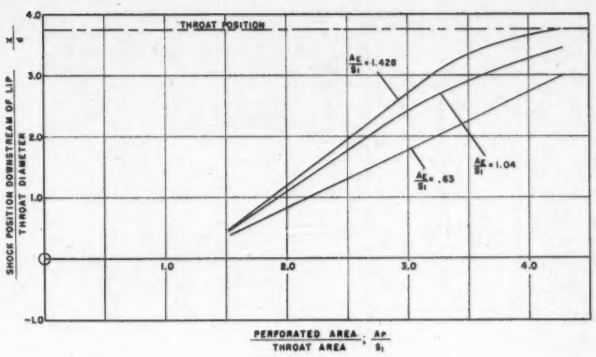

THROAT POSITION
Ae/S1 = 1.428
Ae/S1 = .63
Ae/S1 = 1.04
SHOCK POSITION DOWNSTREAM OF LIP / THROAT DIAMETER, x/d
PERFORATED AREA / THROAT AREA, Ap/S1

Figure 11
Normal shock position versus perforated area

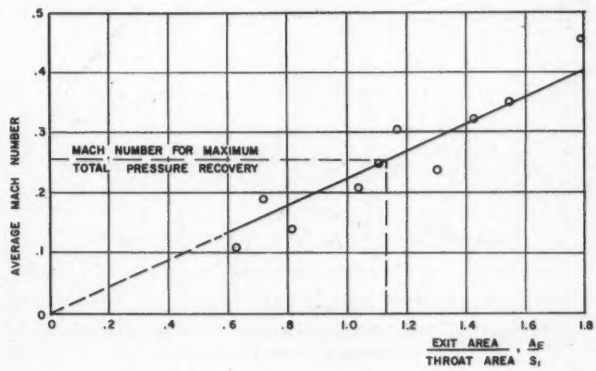


Figure 12
Average Mach number at measuring plane

evidence of a choked throat (i.e. $M = 1$) followed by supersonic flow terminated by the normal shock. The highest exit area ratios force this normal shock to occur further downstream from the throat and cause the system to incur losses approaching those of the unperforated intake.

Figure 10 indicates normal shock positions inside the intake for various amounts of wall perforation at an exit area ratio of 1.428. The intention behind the static pressure tapping of the converging part of the inlet was to obtain an empirical relation between shock position and perforated area. To further this, perforations were drilled in rows at constant intervals of ¼ inch. However, the results do not give a clearly defined shock position due mainly to interference between the shock wave and the boundary layer in the region of the static taps. This results in the growth of a lambda foot at the junction of the shock and the boundary layer, and a comparatively slow increase in static pressure will be obtained rather than the sharp jump in pressure which would be experienced by the two adjacent taps in the ideal case. It is further believed that the radial velocity component of the bleed-off mass flow adds to obscuring the interpretation of the wall static pressure measurements.

Figure 11 shows the approximate variation of shock position with perforated area, as obtained from Figure 10, by arbitrarily assigning the shock to be at the point of maximum slope of the static pressure curves of this figure. At the smaller exit area ratios only, it may be

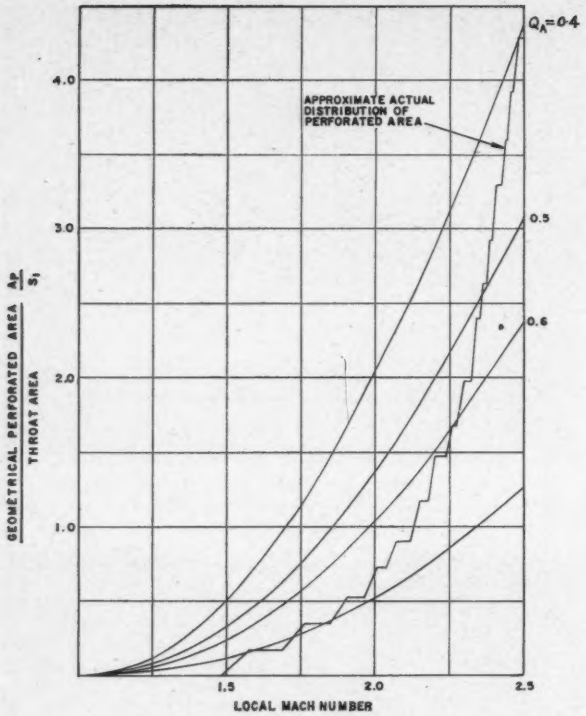


Figure 13
Theoretical and actual perforated area distribution (measured from throat)

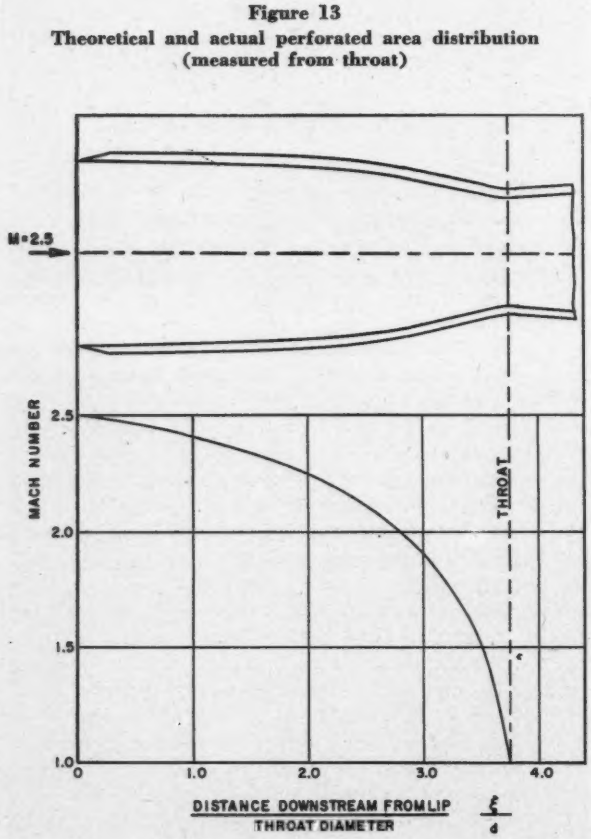


Figure 14
Theoretical variation of Mach number inside intake after starting (calculated from one dimensional theory with no outflow)

said that the variation of shock position with amount of perforated area is approximately linear (Figure 11).

It appears that the variation of the average Mach number at the measuring plane is linear with exit opening (Figure 12) and that at the value of A_E/S_1 for maximum pressure recovery this average Mach number is about 0.25. It might be noted that this value is presently regarded as being about the optimum entrance Mach number to ramjet combustion chambers using hydro-carbon fuels.

Figure 13 shows as an ordinate the theoretical and an approximation to the actual distribution of perforated to throat area ratio along the intake. The abscissa of this figure is the Mach number at any station inside the converging part of the inlet at which the normal shock is stationed. The value of this Mach number is found by applying one-dimensional isentropic flow theory from the lip to the station being considered. Its variation along the intake is shown in Figure 14. The theoretical distribution of perforated area is shown for several values of the discharge coefficient Q_A for the holes downstream of the shock. The method of calculating the theoretical distribution of perforated area is as given in Reference 3.

Figure 15 gives the theoretical variation of the equivalent cross-sectional area of the intake, taking into account the flow through the perforations with the discharge coefficient as a parameter. The basis of obtaining this cross-sectional area is laid down also in Reference 3.

The loss in mass flow through the perforations may be estimated (from the method of Reference 3) by the following relation.

Percentage loss =

$$\frac{(\text{Equivalent Area Ratio at } M = 2.5 - \text{Isentropic Ratio}) \times 100}{\text{Isentropic Ratio}}$$

Using a mean value of Q_A equal to 0.5 the percentage loss is

$$\frac{3.249 - 2.637}{2.637} \times 100 = 23.25\%$$

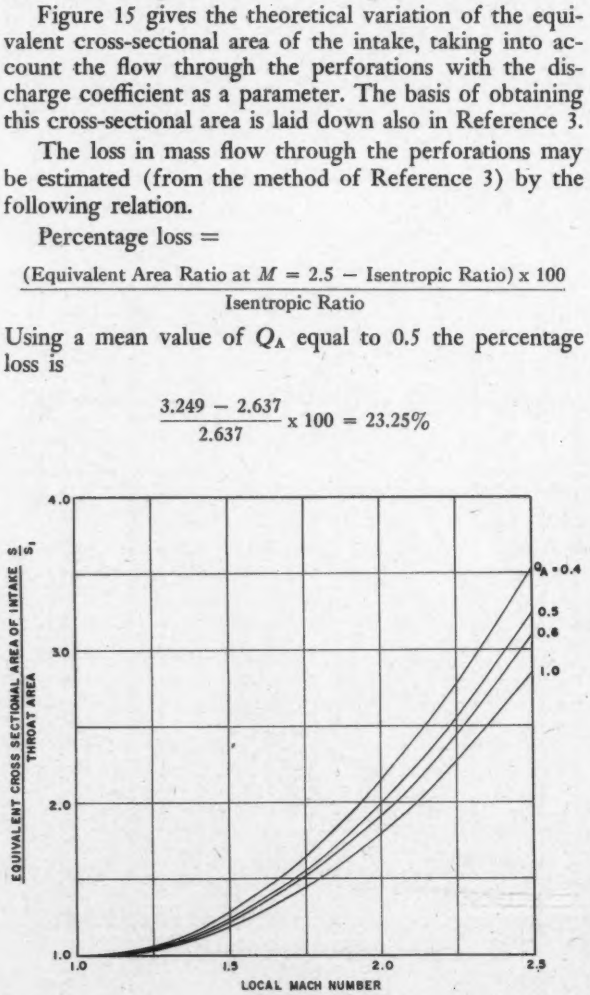


Figure 15
Cross sectional area of intake versus Mach number

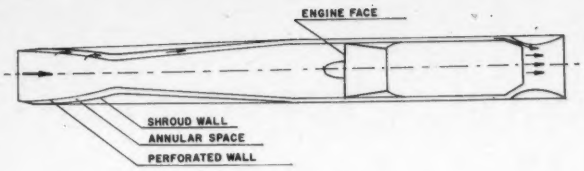


Figure 16
Engine ejector scheme

CONCLUSIONS

It has been shown that a high value of total pressure-recovery (81%) is possible with a perforated wall intake diffuser at a Mach number of 2.5 but only at the expense of mass flow. It is imperative to retain all of the intake mass flow, since the loss in thrust due to a mass flow of 23% is greater than the loss in thrust that would be incurred by an intake having a pressure recovery of 61% and no mass flow loss, i.e. a single 30° cone external compression inlet running critically at maximum mass flow (normal shock on lip).

Furthermore, the external drag of a perforated (internal compression) intake venting to atmosphere might approach or exceed that of a spike inlet (external compression) diffuser. A gain might be obtained if all the bleed air could be shrouded from the external flow and taken away to the rear with the aid of an ejector (see Figure 16). However, this might involve losses owing to the possible formation of shock waves or separated flow in the annular space between the shroud and intake wall.

Another possibility is that of closing the perforations once the intake has been started and the normal shock is downstream of the throat. A slight increase in flow area from lip to throat would be necessary to allow for the thickening of the boundary layer after the holes ceased to discharge, or some method of sucking away the boundary layer through the holes must be implemented if the flow area is not increased. A simple mechanical arrangement, at least suitable for models, is shown in Figure 17.

There are some inherent advantages, however, for a simple perforated intake. Firstly, owing to the high pressure recovery, absolute pressures encountered in an engine or ramjet combustion chamber would be higher than those for a spike inlet engine. Secondly, the perforations act as automatic valves so allowing the positioning of the shock anywhere in the intake to suit back pressure, i.e. engine mass flow requirements, and, thirdly, unstable flow at a flow Mach number of 2.5 is only encountered at small exit openings.

To conclude, it might be said that a simple perforated intake venting to atmosphere does not offer an advantage in pressure recovery great enough to offset the loss in intake mass flow. But if this bleed-off mass flow is compensated for by a correspondingly larger inlet cross-section, the perforated wall internal compression intake seems to be the most promising for higher supersonic flight Mach numbers.

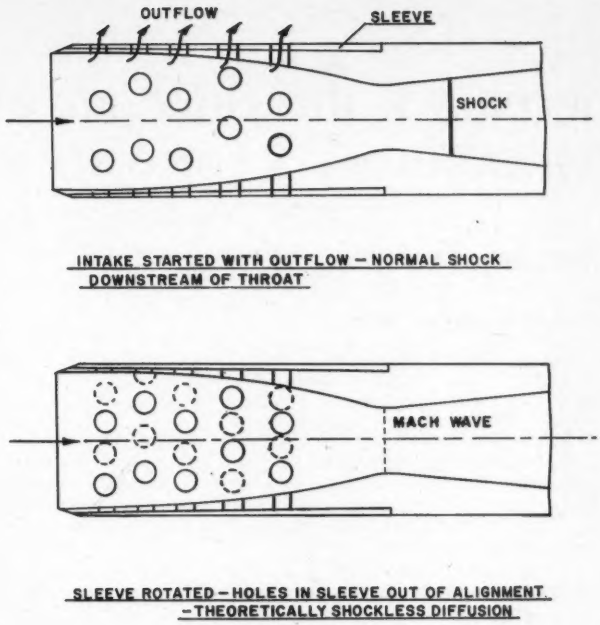

OUTFLOW
SLEEVE
SHOCK
INTAKE STARTED WITH OUTFLOW – NORMAL SHOCK DOWNSTREAM OF THROAT
MACH WAVE
SLEEVE ROTATED – HOLES IN SLEEVE OUT OF ALIGNMENT – THEORETICALLY SHOCKLESS DIFFUSION

Figure 17
Scheme for closing perforations after starting

ACKNOWLEDGMENT

This work was carried out at the Institute of Aerophysics, University of Toronto, under the general direction of Dr. G. N. Patterson.

The author is indebted to Dr. Patterson for suggesting this problem and to the staff of the Institute of Aerophysics, particularly Dr. G. K. Korbacher for his supervision, helpful advice and criticism.

The work was made possible through the financial aid of Orenda Engines Limited.

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WILL WE BE ABLE TO OPERATE A TURBOJET TRANSPORT ECONOMICALLY?†

by C. H. Glenn*

Trans-Canada Air Lines

SUMMARY

The author outlines some of the problems facing the turbojet transport operator with respect to actual flying and despatch techniques. The significance of such variables as speed, ambient temperature, altitude, wind, weight and fuel price are brought forth and the effect on the operating cost of their variation from the optimum is discussed.

The Conway powered Douglas DC-8 aircraft now on order by Trans-Canada Air Lines has been used as a basis for this paper. However, the author implies that operators of other turbojet transport types may be faced with similar and perhaps more complex problems.

The author feels that for some airline operators, the electronic computer may be the answer to solving the complex flight despatch and cruise technique problems; however, he does not close the door to solving these problems by conventional methods.

INTRODUCTION

WILL we be able to operate our turbojet transports economically? The answer is yes, providing we discard all our previous ideas about flying piston powered airplanes economically and start afresh, taking into consideration all the factors and variables peculiar to our particular turbojet.

The actual flying technique and the adequacy of meteorological forecasts, plus other virtually uncontrollable factors, will to a major degree determine whether or not the various operators will be able to show a substantial profit in the operation of their Douglas DC-8's and Boeing 707's. If we assume that the traffic is available to keep our airplanes loaded to the desired load factors and that the revenue per passenger mile is reasonable, then the profit realized will largely depend on our ability to fly the airplanes economically.

OPERATING COSTS

For a fleet of six turbojet aircraft, such as the DC-8, it is expected that the annual direct cost of operation will be in the neighbourhood of $20,000,000. This does not include all those items normally considered as indirect costs, i.e. buildings and ground equipment, advertising, sales, administrative etc. For some of the larger U.S. airlines, with many more of these giant jet transports on order than has TCA, the annual direct operating cost could be around $150,000,000. A slight variation in this operating cost could mean a substantial profit or loss. The smart airline operator will be the one who knows where the direct operating costs are being spent, and tries to do something about minimizing them.

If we disregard items which are not affected by flying technique, such as depreciation and insurance, landing fees etc, we find that fuel will make up about 50% of the actual flying cost. The remainder is broken down between maintenance, labour and material, crew salaries etc, which can be shown as a fixed cost per hour, more or less independent of speed and range. This is not strictly correct since items such as engine material costs vary with the amount of thrust being developed and, theoretically, the faster you can fly the more trouble you can expect with the aircraft structure. However, with the slight variations in speed considered in this study, the effect is negligible and it will be assumed that all these direct costs, other than fuel, vary directly with the takeoff to touch down time.

DRAG RISE

As compared with other machines doing useful work, the turbojet engine is quite inefficient. As long as we have to use this method of propulsion (and we know of nothing better for the subsonic transport) we will be faced with high fuel consumptions, and high fuel consumptions mean high fuel costs.

All the blame cannot be placed on the engine since the airplane itself shares a good deal of the responsibility for high fuel consumption and hence high costs. Figure 1 shows the relationship between airplane drag and speed for the DC-8 at a fairly high weight and high altitude. If we remember that drag equals thrust for level flight, and to produce thrust we require a fuel flow, then this curve could just as easily have been drawn as fuel consumption against speed. At this particular altitude, an increase of speed of 10% from say Mach 0.8 to Mach 0.88 would increase the fuel consumption by 50%. This could, under certain conditions, cut into payload and at the same time would most certainly increase the cost of operation.

The sharp rise in the airplane drag as the speed is increased results from the effects of compressibility. The effects of compressibility can be delayed by increasing the wing sweep and making it thinner. However, once an airplane is built and delivered, there is little the airline operator can do about its characteristics.

CRUISE CONTROL

The primary objective in any cruise control technique is to permit the carrying of as much payload as

†Paper read before the Winnipeg Branch of the C.A.I. on the 24th June, 1958.

*Project Engineer — Aircraft Evaluation.

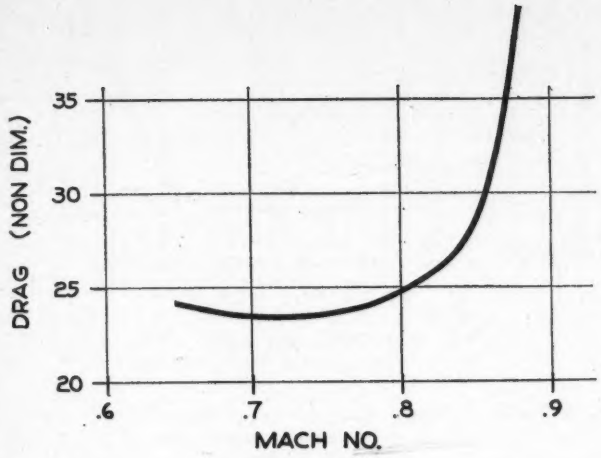


Figure 1
Drag rise — DC-8
High weight — high altitude

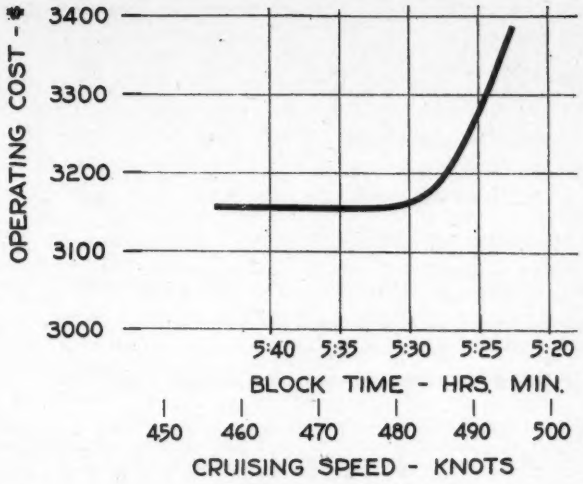


Figure 2
Effect of cruise speed
Stage length 2,500 nautical miles
Altitude 35,000 ft

possible. After all, airline revenue comes from passengers, cargo and mail, and it just doesn't make sense if we use the most economical cruise technique from the point of view of operating costs, but lose out on payload by doing so. Once we find that we can carry the full payload on any route leg *then* we can concentrate on the most economical techniques. As a general rule, however, any procedure which saves fuel results in the lowest operating cost. Of course, there are exceptions to this.

There are many variables to be considered in deciding on a cruise technique for a particular operation, all of which affect the cost of operation and many of which affect the range and payload. Some of these are speed, temperature, altitude, wind, aircraft weight, price of fuel etc. The best technique for flying a given route one day may be entirely different the next. Let us look at the effect of a few of these variables.

For all the curves shown, unless otherwise noted, the airplane is the DC-8 with Conway engines at standard temperatures, a given altitude, a given landing weight, a range of 2,500 nautical miles with normal reserves, a given fuel price and a given cost per hour for those items of direct operating cost which have a fixed rate per takeoff to touch down hour.

EFFECT OF SPEED

Figure 2 illustrates the effect of speed on the total cost of flying this particular route. Note the sharp increase in costs above 480 kts. The block time in this example for 2,500 nautical miles at 480 kts is about 5 hrs, 31 min. If we were to increase the cruising speed to 495 kts, i.e. by about 3%, the cost of flying this route would increase by about 7%. The reduction in block time would be about 8 min, a gain hardly worth the additional cost.

If payload were critical on this route, flight planning for a speed higher than the optimum could mean a considerable loss of revenue. Had the flight been planned for the optimum speed but a higher speed used in the actual operation, the cost of operation would have gone up and the pilot would have cut into his reserves.

In this example, to fly slower than 480 kts would offer no advantage. In this region, any savings due to lower fuel consumption are nullified by the increase in block time.

EFFECT OF TEMPERATURE

Before discussing the effect of temperature, it will be necessary to back track a bit and discuss a little theory.

One of the methods of classifying turbojet transports is by the factor which limits the economical cruising speed. For the coming generation of turbojet transports we will see both thrust limited and drag limited airplanes.

A thrust limited airplane is one which, for standard day cruising, requires that the engines run at their maximum cruise thrust for the most economical operation of the airplane. With this type of airplane, anything which reduces the maximum available cruise thrust, such as an increase in ambient temperature, results in the loss of airplane altitude and/or speed, with an associated increase in the cost of operation and perhaps a reduction in available payload. The Douglas DC-8 and the Boeing 707 airplanes powered with the Pratt & Whitney J-57 fall into this class.

A drag limited airplane is one which has its most economical operating point slightly up on the drag rise curve with the engines normally running at something less than the maximum cruise rating. This class of airplane is over-powered for standard day cruise operation and can suffer a reduction in the allowable maximum cruise thrust when operating in ambient temperatures above standard, without penalizing the performance and hence the economics of the airplane. The Douglas DC-8 and Boeing 707 airplanes powered with the Pratt & Whitney J-75 and the Rolls-Royce Conway engines fall into this class.

A drag limited airplane, although initially more costly than a thrust limited airplane due to the inherently larger engine, is usually more flexible from the point of

	STANDARD		STANDARD+20°C
ALTITUDE - FT.	35,000		35,000
MACH NUMBER	.825		.825
SPEED - KNOTS	475	→	496
THRUST AVAILABLE - LBS.	5000		4000
THRUST REQUIRED - LBS.	4000		4000
FUEL FLOW PER ENGINE - LBS./HR.	3500		3640
AIR MILES PER POUND $= \frac{\text{SPEED}}{\text{TOTAL FUEL FLOW}}$	.0339	→	.0341

FUEL FLOW & THRUST ARE APPROX. AS CONWAY DATA RESTRICTED

Figure 3
Effect of temperature on speed and miles per pound

view of air traffic control and last minute changes in flight planning. At the same time, the use of this class of airplane virtually eliminates cruising temperature as a factor to be considered in the advance sale of seats to the public.

Fortunately, the DC-8 becomes drag limited with the Rolls-Royce Conway engine installed, and even though an increase in cruising temperature reduces the maximum thrust available from the engines, the net result is a higher cruising speed, and slightly more miles per pound of fuel for temperatures up to about standard +20°C. This will be explained further, later on.

Another point to remember is that the speed of sound is dependent only on temperature. The higher the temperature, the higher the speed of sound. Since Mach number is the ratio of the true airspeed to the speed of sound, it follows that for a given cruising Mach number, the higher the temperature, the higher the true airspeed.

The last basic item to remember is that for a given aircraft weight and altitude, the drag of an airplane (and hence the thrust required) is dependent on the Mach number.

Let us apply this theory in an example comparing the speed and miles per pound of fuel for a given drag limited airplane under standard conditions with standard +20°C for a particular weight and altitude. The calculations are shown in Figure 3.

Note how there has been a gain in speed and a slight improvement in air miles per pound as the temperature increased. Had this been a thrust limited airplane, it could not have held 35,000 ft at the higher temperatures and the fuel consumption, and perhaps the speed as well would have suffered.

In this example the maximum allowable thrust and the specific fuel consumption figures have been changed from the actual values, since the Conway engine performance is still restricted. However, for the correct values, the relationship still holds.

Applying the above type of calculations to the DC-8 cost curves gives results as shown in Figure 4. The upper curve is the same as in Figure 2. The lower one is for the elevated temperature. The diagonal lines are those for constant Mach number.

Note that the increase in temperature has shifted, to a higher speed for the same Mach number, the rise in the curve which is due to the effects of compressibility.

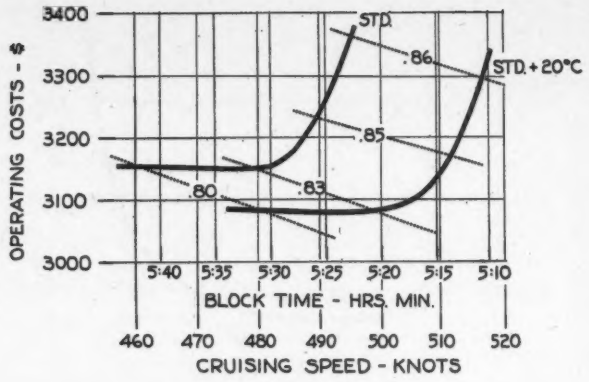


Figure 4
Effect of cruise temperature
Stage length 2,500 nautical miles
Altitude 35,000 ft

The curve shifts down as well, due to the effect of the higher speeds on the fixed costs and also due to the slight improvement in fuel consumption.

The improvement due to higher temperatures can be used to permit higher speeds (lower block times) for the same operating cost as for standard temperatures, or can be used to give lower costs for slightly higher speeds. Since airline timetables are based on standard temperatures, little would be gained from the point of view of passenger appeal etc by cruising at any Mach number higher than 0.83 in this case. The smart operator will take the advantage in the form of a reduction in operating cost since, unfortunately, a reduction in temperature below standard has an adverse effect. What is saved in the warm summer weather is lost during the cold winter.

This illustrates the importance of accurate and up to date meteorological information. It may be desirable for a pilot to deviate to the warmer regions to improve the speed and/or economics. "Temperature pattern" flying may prove advantageous.

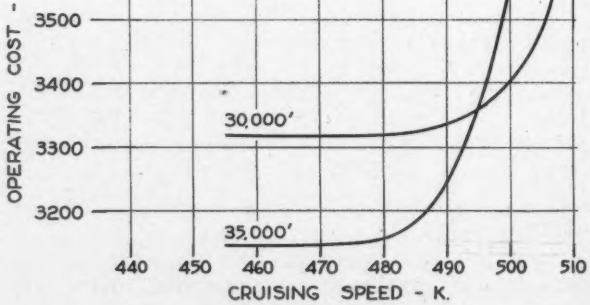


Figure 5
Effect of cruising altitude
Stage length 2,500 nautical miles
Standard conditions

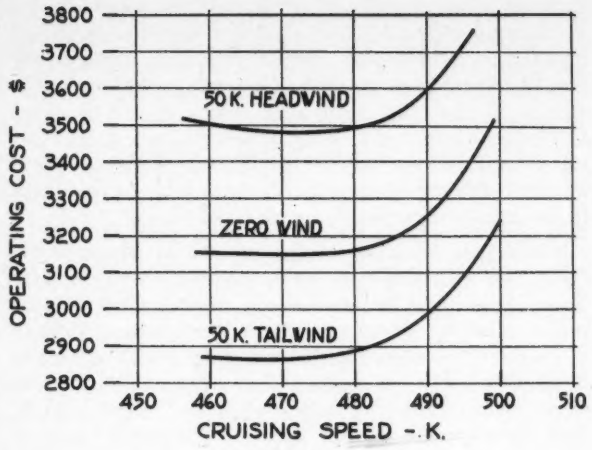


Figure 6
Effect of cruise wind
Stage length 2,500 nautical miles
Standard conditions

EFFECT OF ALTITUDE

Altitude has perhaps a more marked effect on flying costs than any of the other variables. This is shown in Figure 5. Note that the operating cost increases by about 13% in dropping from 35,000 ft to 25,000 ft for say 480 kts. Between 30,000 ft and 35,000 ft there is a change of about 1% per thousand feet.

In this figure, note that as the altitude decreases the upswing in costs is delayed slightly. This is due to (a) the increase in the speed of sound due to the higher temperatures at the lower altitudes and (b) the lower angle of attack for a given speed and weight, i.e. a lower lift coefficient. The drag rise up to a certain point on the DC-8 is lower at low lift coefficients.

EFFECT OF WIND

Figure 6 shows the effect of wind on the operating costs. Unlike slower piston powered airplanes, there appears to be no advantage economically to speeding up when flying against a headwind.

For this particular stage length, there is roughly a 1% change in the cost per 5 kts. Referring back to Figure 5, note that between 30,000 ft and 35,000 ft there is about a 1% change in costs per 1,000 ft due to basic fuel consumption. One would gather then that if the head wind increased with an increase in altitude at a rate greater than 5 kts per 1,000 ft, it would be faster and more economical to stay lower. This applies only to this 2,500 nautical mile stage length and a variation from this range, or a change in ambient temperature, will have the effect of varying this 5 kts per thousand feet.

EFFECT OF WEIGHT

Another variable is, of course, the weight of the airplane less the route fuel, i.e. the landing weight. The overcautious pilot who carries a few thousand pounds of fuel "for the wife and kids" over and above the required reserves increases the operating cost, since it costs fuel to fly around the extra fuel. This is shown in Figure 7. The effect is slight, but it all adds up.

EFFECT OF FUEL PRICE AND TYPE

Of major importance is the price of fuel. Figure 8 shows how one or two cents a gallon will affect the economics. For example, one operator using one type of fuel costing, let us say, 20 cents a gallon would have to fly at about 480 kts for this set of conditions for best economy, whereas another operator using a fuel say 2 cents cheaper could afford to fly 15 knots faster, for the same operating cost. This could be reflected in published timetable times and have a marked effect on the travelling public. Other factors being equal, the public go on the faster airline. To compete, the airline using the 20 cent fuel would have to speed up, which would increase his operating costs considerably.

Independent of fuel price, fuel type could have a marked effect on the operating economics. At ranges where the amount of fuel carried cuts into the payload, certain fuels have a considerable advantage. Figure 9 shows the difference between JP-4 and kerosene with

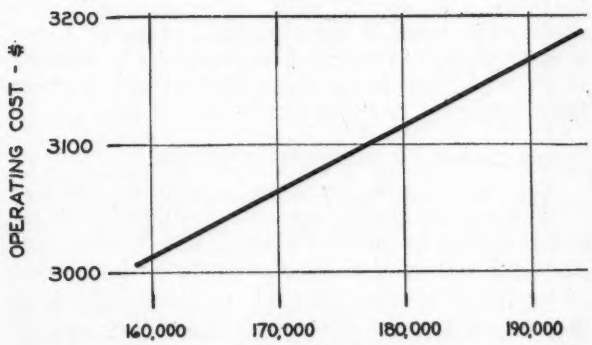


Figure 7
Effect of landing weight
Stage length 2,500 nautical miles
Standard conditions
Mach 0.84

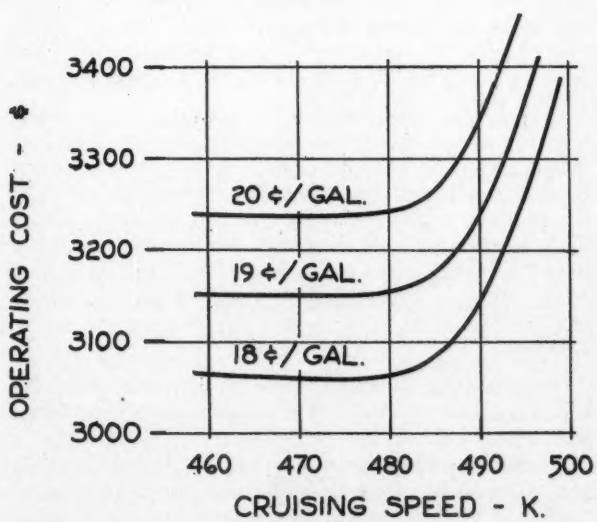


Figure 8
Effect of fuel price
Stage length 2,500 nautical miles
Standard conditions

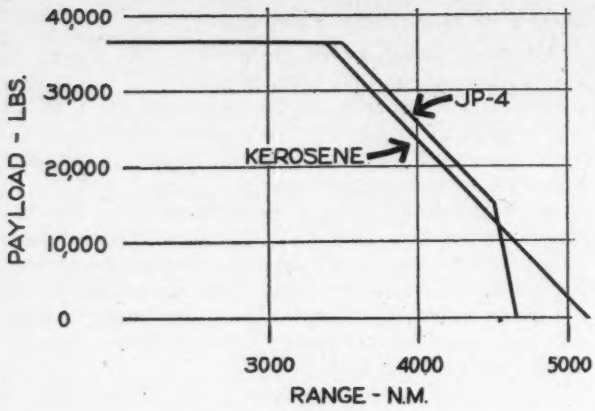


Figure 9
Effect of fuel type on range
Douglas DC-8

respect to the payload and range. For ranges between 3,500 and 4,500 nautical miles, the airplane using JP-4 could carry 1,600 lb more payload, or about 8 passengers. Beyond 4,500 miles kerosene would be superior. However, at these ranges the payload is so low that the flight would not be economical in the first place.

CLIMB CRUISE VS CONSTANT ALTITUDE

We have seen that, in order to keep the operating costs down, we must in general fly as high as we can and at the proper speed. There is a limit to which this can be carried, as shown in Figure 10.

To the left, the straight lines for heavy, medium and light weights indicate the minimum speed for acceptable controllability. As would be expected, the lower the weight, the lower the speed or Mach number for controllability. Intersecting these lines are the curves of "Drag Divergence". Those are arbitrary limits set by the airframe manufacturer, beyond which the fuel consumption is considered by them to be excessive, resulting in uneconomical operation. These are not speed limits, since the structure would be satisfactory for cruising up to the V_c limit shown on the extreme right.

These drag divergence curves are supposed to represent the points where the drag rise due to the local velocities on the airplane exceeding the speed of sound become significant. Since the local velocities vary with the angle of attack, and since the angle of attack varies with weight (for a given speed), we find that there is a drag divergence curve for each weight. These are shown intersecting the corresponding lines for minimum controllability for the airplane at light, medium and heavy weights.

If we cannot fly slower than the controllability limit and should not fly faster than the drag divergence limit, and want to fly as high as possible, the place to fly is the intersection of the two. The area just below the peak formed by these two curves is sometimes called "optimum corner". In this area, if you slow down slightly, you become unstable; if you speed up, your fuel consumption suffers.

Long range cruise procedures recommended by the airframe manufacturer call for flying up "optimum corner". In other words, as you get lighter, due to the fuel burnt off, you climb to a higher altitude in order to derive the benefits of lower fuel consumptions. This procedure is probably acceptable where traffic is light. However, we do not feel that it will be permitted across the North Atlantic or in the U.S. or Canada where constant altitude or step climb flight will undoubtedly be required for air traffic control reasons.

With the knowledge that we would probably be faced with restrictive air traffic control procedures, we selected an engine for our DC-8's, the Rolls-Royce Conway, which would allow us to fly a constant altitude technique as economically as a climbing cruise technique with the competitive engine. This constant altitude, for standard conditions, is higher than the initial altitude used for the same airplane with the competitive engine. This results from the combination of the following:

(a) For a given stage length, the DC-8 with the Conway engine is lighter (due to the engine being lighter and a slightly lower fuel consumption), enabling the airplane to start its cruise at a higher altitude than the airplane equipped with competitive engines.

(b) Since drag divergence is a theoretical limit arbitrarily set by the airframe manufacturer, based on their opinion of the upper economical speed limit, by using an engine which gives a lower fuel consumption, speeds above the drag divergence can be justified.

(c) By using an engine, such as the Conway bypass, which, for a fixed throttle setting, has a slightly decreasing thrust with increasing speed rather than increasing thrust with increasing speed for a conventional turbojet engine, a slightly lower minimum speed can be maintained for controllability for a given weight.

The combined effect of the above results in a lifting for a given stage length of the initial "optimum corner". Starting off cruising in "optimum corner" and then cruising constant altitude at a constant Mach number, the controllability of the airplane becomes better as the airplane becomes lighter.

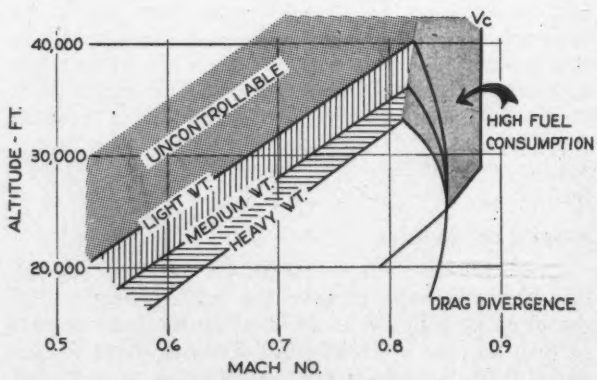


Figure 10
Cruise limits

The effect of this procedure on range and payload is shown in Figure 11. It must be remembered that this figure compares the airplane with the same engines, using the two techniques. As mentioned above, the range capability of the DC-8 with the Conway engine, using the constant altitude technique, is as good as the DC-8 with the competitive engine, using the climbing cruise technique.

The penalty which we would pay by cruising at a fixed altitude would be a loss of potential range of about 50 to 80 miles, which is not significant when the advantages are considered. This loss could be minimized by reducing speed as the airplane becomes lighter; however the airplane would again be operating near the limits of controllability, and it is doubtful that the pilot would want to fly a long flight in this manner.

SUMMARY OF VARIABLES

To sum up then, there are many variables which should be considered in deciding on a cruising technique for a particular flight. All of these factors affect the operating costs and in many cases the payload and range. A summary of these is shown in Figure 12. Every one of these affects the other to some degree and there are no simple rules that can be laid down to allow a despatcher or pilot to decide on the optimum technique.

An accumulative combination of the variables working in an adverse direction could conceivably push the operating cost up by 10% or more. By the use of "operating manual" or "chart" types of techniques, as we use today, we could probably achieve an average operating cost of about 5% from the optimum. Unfortunately, there are more variables working "against" you than "for" you and one should not take the attitude that where today's flight is costly, tomorrow's will be economical. If all the variables are not considered and applied to both the despatch and cruise techniques, the net result will be a more costly operation.

USE OF COMPUTERS

One of the answers to solving the daily complex problems, which will most certainly arise in this area, is the electronic computer. These are being used now by airframe manufacturers for forecasting performance and economics and can be easily applied to airline operation.

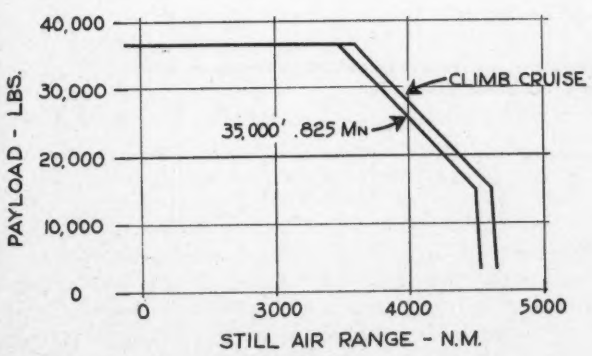


Figure 11
Range payload — normal reserves
Douglas DC-8

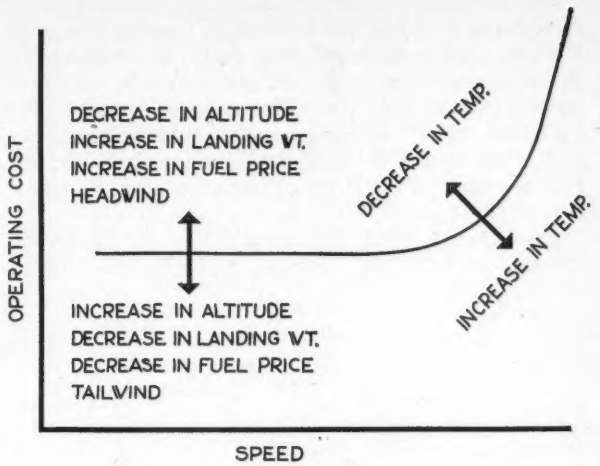


Figure 12
Factors affecting operating cost

The data fed into the computer would be:

(a) anticipated landing weight of the airplane,
(b) stage length (or various track distances),
(c) forecast en route variations of wind and temperature with altitude and track,
(d) price of fuel,
(e) type of fuel,
(f) fixed cost per hour, and
(g) basic performance of the airplane-engine combination.

The information turned out by the machine would be:

(a) best altitude and speed (or any changes in these if en route conditions dictate it),
(b) best track,
(c) fuel to be boarded,
(d) payload which can be carried,
(e) block time, and
(f) operating cost.

If ATC would not permit the procedure recommended by the computer, then the allowable procedures could be fed back into the computer and the best one used.

The small operator with one or two of these jet airplanes probably could not justify the large expenditure for an electronic computer. With only a small technical staff and probably less detailed meteorological data available he will find it hard trying to make ends meet unless he can maintain a relatively higher load factor, operates on a route with relatively high rates, or achieves economies in some other section of his company.

The large operator could probably afford a computer. At least one of the large North Atlantic operators is planning on this method of despatch and cruise control.

The middle sized operators, who probably cannot afford a computer for flight planning, will have to work from information contained in voluminous manuals.

Using such manuals, and considering the number of variables to be taken into account, it is unlikely that all but the more important factors could be considered in the short period of time available before a flight for making out the flight plan. This may not mean that the operation would not be safe or would be extremely costly, but simply that, by not considering all the factors, the operator could not expect to achieve the lowest cost of operation.

Since actual winds and temperatures do not always correspond with the forecast, perhaps consideration should be given to the use of simple airborne computers to assist the flight crew in changing the flight plan if conditions dictate it.

CONCLUSION

Flight planning and operating turbojet transports to allow the maximum payload to be carried at a minimum of cost of operation present a challenge to the airlines. The airlines must start now with programs of indoctrination so that they will be in a position to meet the challenge.

McCURDY AWARD

The McCurdy Award will be presented at the Annual General Meeting which will be held on the 15th-17th June, 1959.

It is presented each year

To A Resident of Canada,

For Achievement in design, manufacture or maintenance related to aeronautics.

NOMINATIONS ARE INVITED

Each nomination should include

(a) The name and affiliation of the nominee,

(b) Confirmation that he is a resident of Canada,

(c) A citation of the particular achievement for which the nomination is being put forward, and

(d) The name of the nominator.

The nominee need not be a member of the C.A.I. and the achievement need not have taken place within the last year, though it should be recent.

Nominations should be in the hands of the Secretary not later than the 14th March, on which date they will be handed over to the McCurdy Award Selection Committee.

C.A.I. LOG

SECRETARY'S LETTER

1959 — COMMEMORATIVE POSTAGE STAMP

The National Coordinating Council for the Golden Anniversary of Flight in Canada, of which the Institute is a member organization, has arranged for the issue of a special postage stamp on the 23rd February, 1959. Special "first day covers" will also be available. No doubt our philatelically-minded members will know what to do about these things but, if they want any additional information, I hope they will let me know and I will try to obtain it.

OVERSEAS BRANCHES

With Canadian Forces stationed in Europe, there is inevitably a fair sprinkling of members of the C.A.I. in France and Germany. And there are members living in England too. From time to time someone puts forward the idea that we should form a European Branch or a British Branch.

Firstly, I must say that the enthusiasm of those who have put forward these suggestions is most impressive and altogether commendable. But I doubt that they appreciate what a Branch really is or what its organization entails. I believe that what they really want is something less formal; in this we may be able to help them but, as I see it, we cannot encourage the formation of Branches outside Canada.

The first requirement for the formation of a Branch is that there should be at least twenty members at some "centre of aeronautical activity"; I think that it is implied that the members must live near enough to that "centre" to enable them to participate regularly in Branch activities. A Branch is required to hold at least four technical meetings each year. And the Council has established certain minimum requirements about the Branch Officers and the manner in which elections shall be conducted. Perhaps most significant of all, in considering the possibilities of an overseas Branch, a Branch is financed by a "kick-back" of $2.00 a year of each member's dues – and overseas members enjoy a reduced rate primarily because they are unable to take part in Branch activities.

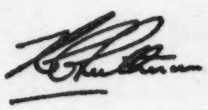
For these reasons, it is impracticable to organize Branches overseas. Furthermore in the United Kingdom the R.Ae.S. holds sway and can give far better service to members living there than we can ever hope to give. It is quite unthinkable that we should try to compete with them on their home ground, particularly since they have agreed not to compete with us on ours.

However, there is certainly no objection to an occasional, informal get-together by members living overseas, if they want to entertain some eminent visitor from Canada or something of that sort. If such a meeting is ever planned, I shall be glad to supply a list of names and addresses of the members in the country concerned, so far as I know them – some of the people in Europe have only a central Service address in Montreal. And of course we will publish any report on the affair that may be sent in to the Journal. But that is about as far as we can go.

STAFF

W/C H. L. Taylor, who has been Assistant Secretary for little more than a year, left us at the end of December to take a position in the machine tool industry. Many members got to know him during his service with the Institute and I am sure that I speak for all of them in thanking him for what he has done for us and wishing him well in his new job.

He tells me that he intends to keep up his membership and I hope that we shall see him occasionally at some of our meetings.



BRANCHES

NEWS

Halifax-Dartmouth

Reported by Lt. J. A. Turner

November Meeting

The monthly meeting was held in the cinema of the CPO's Mess, HMCS Shearwater, on Tuesday, 20th November, 1958. A total of 26 members and guests were present.

The Chairman, CPO R. L. Sabourin, informed those present that a letter had been sent to the senior engineering associations in the Halifax area requesting that they participate with the Branch on the 23rd February, 1959, to commemorate the 50th Anniversary of the first successful flight in Canada.

Mr. R. Wallworth of the Programme Committee outlined the details of a Buffet and Dance which will be held on the 12th December, 1958. He pointed out that the assistance of all the members would be required in order to make the evening a success.

The speaker for the evening, Mr. M. C. Eames, of the Naval Research Establishment, was introduced by Mr. E. C. Garrard. Mr. Garrard noted that this was the second paper to be presented by the speaker during the relatively short life of the Halifax-Dartmouth Branch.

Mr. Eames presented a paper on "Some Recent Developments on Hydrofoils". During the initial stages of his talk, he reviewed some of the aspects covered in the paper which he presented in April, 1957. It was noted that the developments which have been made in the cavitation field since that time will permit an advance in the hydrofoil field comparable with that made in the aircraft field with the introduction of the gas turbine. The speaker then traced the development of the hydrofoil from its initial concept up to the super-ventilating foil. Cavitation has been the limiting factor throughout the development of the hydrofoil craft, but with the introduction of the super-ventilating foil it is anticipated that great progress can now be made in this field. The improved hydrofoil design would have an application to both nuclear-powered ships and water-based aircraft.

After a brief intermission, film sequences were shown of some of the tests conducted on hydrofoil craft by the Naval Research Establishment. This was followed by a very interesting and enlightening discussion period. Mr. Wallworth thanked the speaker on behalf of those in attendance.

Vancouver

Reported by G. W. T. Roper

November Meeting

On the evening of November 21, 1958, the Vancouver Branch held its fall dance in the B.C. Air Lines hangar at Vancouver Airport.

Approximately 160 members, wives and guests, dressed in a variety of colourful costumes, wined, dined and danced the evening away in a most enjoyable fashion. The many table favours and door and spot dance prizes donated by various airlines and local companies added to the fun of the evening. The hot meal served at such a remote location was nothing short of excellent. The Program Committee is to be commended on the effort expended on the arrangements which made this affair an outstanding social success.

Edmonton

Reported by P. F. Leigh-Mossley

December Meeting

The fourth meeting of the Branch for the 1958-59 season was held at the RCAF Association, 700 Wing, at 8.00 pm on the 10th December.

Mr. C. C. Young, Branch Chairman, welcomed 16 members and 13 guests to the meeting. The Secretary read the Minutes of the third meeting. It was proposed by Mr. J. G. Portlock and seconded by F/O A. J. Robinson that the Minutes be adopted as read.

The Chairman spoke to the meeting about financial problems and membership and explained that each member in good standing brought $2.00 per annum into Branch funds. He asked all members to make sure that their dues were paid to Headquarters and to introduce new members to the best of their ability. He read the programme of lectures for the season and said that most of the lecturers travelled a long distance to address the Branch and

Scenes of the Vancouver Branch November social evening

could be very disappointed if only a handful of people turned out to listen to what they had to say.

The guest speaker, Mr. H. Whiteman, Project Engineer (Argus) of Canadair Ltd., was then introduced by Mr. C. W. Arnold.

Mr. Whiteman explained the background of the Argus and showed how the decisions to choose the Bristol Britannia were reached. After the project had been started, a Committee was set up with members of the Air Force and Dept. of Defence Production and Canadair Ltd. Sub-committees were then formed to deal with such items as crew accommodation, armament etc, and following this a complete model specification was drawn up and a full-scale wooden mockup was manufactured. This mockup was found to be of tremendous value in the progress of the design. Testing included wind tunnel tests of models, ditching tests and complete reproductions of electrical generating, control and distribution systems. Ground tests were run on the powerplant and antenna radiation measurements were done using three scale models. At the time of the first flight of the prototype aircraft, approximately 256,000 manhours had been spent on tests of various types.

Mr. Whiteman then showed slides illustrating the interior arrangements of the aircraft.

The first aircraft flew at the end of March, 1957, just under three years after the time of initiation of the firm requirement. Mr. Whiteman then described some of the tests which are being carried out on the aircraft and noted some of the differences between the Mark 1 and Mark 2 versions.

In conclusion, he emphasized that the successful completion of the design of the aircraft had been due to very precise and accurate timing of the various stages of design, construction and testing and a strict adherence to time schedules in all sections of the work. The speaker ended his talk with the following quotation from one of the essays of Sir Francis Bacon – "Above all things, order, and distribution, and singling out of parts, is the life of dispatch; so as the distribution be not too subtle: for he that doth not divide, will never enter well into business; and he that divideth too much, will never come out of it clearly. To choose time, is to save time; and an unseasonable motion, is but beating the air."

After a brief recess, the meeting reassembled to see a sound film entitled "Birth of a Giant". Following the film, there was a short discussion period and the speaker was thanked by Mr. J. G. Portlock. The meeting then adjourned.

Ottawa

Reported by R. L. Wardlaw

December Meeting

W/C E. E. McCullough was the Chairman of the meeting on the 10th December. Serious competition from the TV presentation of the light-heavyweight championship fight in Montreal resulted in the low attendance of only 25 members. The attendance was also hampered by uncooperative weather. The delinquent members missed two interesting discourses in the field of aviation medicine.

The two speakers were Surgeon LCDR (P) H. D. Oliver, Principal Medical Officer, RCN Air Station Shearwater, and W/C D. O. Coons, Director of Aviation Medicine, AFHQ. LCDR Oliver was introduced by CDR N. A. Smith and W/C Coons by W/C McCullough.

LCDR Oliver's talk on "Toxilogical Effects of Aviation Fuel" was a practical, instructive outline of the perils associated with the handling of aviation fuels. It was pointed out that of the conventional fuels, gasoline was the most hazardous since some of the components are toxic or poisonous and can lead to disorders of the nervous system, the kidneys or the liver. It was made clear that it is difficult to put numbers to this problem since susceptibility varies with individuals and also with their degree of stress or fatigue.

Whereas LCDR Oliver's address kept us very much down to earth, W/C Coons had us very high in the air with his talk on "Biological Hazards of Space Flight". The problems of high acceleration, weightlessness, isolation over prolonged flights, radiation and waste disposal, along with other things, were discussed. He reported that there was no biological problem associated with weightlessness and that the acceleration of a possible space flight would present no serious difficulties. For a prolonged flight, the sensory deprivation due to isolation and the dispesal of wastes were considered important problems.

Upon the conclusion of a profitable discussion period, the speakers were thanked by A/C A. A. G. Corbet.

Cold Lake

Reported by WO2 J. W. Day

December Meeting

The meeting was held in the CEPE Canteen and 29 members and guests were present. F/L L. S. Lumsdaine was in the Chair and introduced the speaker of the evening, Mr. H. H. Whiteman of Canadair Ltd. The speaker started his talk with references to some of the older type aircraft, such as the Stranraer Flying Boat, and the requirements of the maritime reconnaissance role. The extensive problems associated with the design and building of an aircraft such as the Argus were illustrated by slides and films which the speaker explained fully. His address was followed by a discussion and question period.

W/C W. N. Hoye thanked the speaker on behalf of the Branch.

Montreal

Student Section

On the 24th November the Montreal Branch held a meeting of Student members of McGill University and Sir George Williams College for the purpose of forming a Student Section and electing Officers. P/O A. R. Baker of McGill was elected Chairman.

At the formation of the Montreal Branch Student Section: (l to r) Mr. R. J. Conrath (Student Activities Chairman), W/C C. R. Thompson (Branch Chairman), P/O A. R. Baker (Section Chairman), G/C C. W. Crossland (Student Advisor)

Winnipeg
Reported by G. Milner

December Meeting

The third meeting of the 1958-59 season was held on the 2nd December in the Winnipeg Flying Club dining room. Due to the very bad weather, only 16 members attended. Chairman of the meeting was Mr. G. Milner, who introduced and thanked the speaker. Due to the number of members present, the meeting was conducted strictly on an informal round-table basis.

The speaker was Mr. Ross Screaton, President of the Winnipeg Flying Club, who has been actively associated with flying in the district since 1920.

In opening his talk, Mr. Screaton delved back into the history of flying in Winnipeg and he recalled visits in 1910 to 1916 of such famous stunt and exhibition pilots as Ely, Lincoln Beachey, Frank Coffin, Jim Ward, Bill Straith, Katherine Stinson and many others, all of whom introduced the airplane to this district through the medium of exhibitions and air shows.

The first commercial endeavours were started at the end of World War I when a number of the ex-Royal Flying Corps pilots attempted to use the airplane as a means of transportation. Most of these men, using military training aircraft moved into the barnstorming business, flying passengers on short hops of approximately 10 minutes duration at $10 a head. As the airplane became more common and the novelty wore off, all but a few of the more hardy were forced into other lines of business. A number of aircraft companies were started during this era, all of which disappeared in the bankruptcy courts. A few of the more determined men continued to eke out a living barnstorming at country fairs, doing the odd mercy flight and commercial transportation work. It was not until 1928 that a major commercial operation was attempted under the sponsorship of James Richardson who formed Western Canada Airways, later Canadian Airways. In 1930, this company expanded its operations by introducing the Prairie Airmail. 1928 also saw the formation of the Winnipeg Flying Club under the sponsorship of the Federal Government. The Winnipeg Flying Club dates from that year. The first equipment and facilities consisted of Gypsy Moth and Avro aircraft, operating from a landing area staked out in the approximate location of the present club grounds. The success of this venture was only minor and during the early days of the depression it was forced almost out of existence. In 1933 the need for pilot training became evident as a national requirement and the clubs were reformed and equipped with more up-to-date machines. Government assistance in obtaining a private license was also granted.

Just prior to the second World War, a further government-sponsored expansion took place when it became necessary to train pilots for the growing RCAF. Classes of pilots were trained in Winnipeg for both the RAF and the RCAF. After the declaration of war, private flying was stopped with the exception of commercial pilot testing.

Mr. Screaton dealt at some length with the major contributions made by the flying clubs of Canada during the war in the operation of Elementary Flying schools. The saving in training costs resulting from their management permitted major contributions to be turned back at the end of the war. Since the war, the progress of the Club has steadily moved forward to the point where it now occupies first or second position in Canada in the number of flying hours and membership. The contributions made by this Club to aviation in Canada have been many and great. From this membership have come many of the well-known names in aviation. Its record in training, hours flown and number of graduates and members holds a spotlight in the Canadian Flying Training field.

Calgary
Reported by H. E. Hampshire

December Meeting

A Dinner Meeting was held on Friday, 12th December, at the Provincial Institute of Technology and Art. 34 members and guests were in attendance.

Mr. W. A. B. Saunders opened the meeting by introducing the members and guests at the head table. Following the reading and adoption of the minutes of the previous meeting, a short business discussion followed.

The guest speaker of the evening, Mr. H. H. Whiteman, CL-28 Project Engineer of Canadair Ltd., was then introduced by Mr. W. E. Jamison. Mr. Whiteman chose as his subject the development of the "Argus" aircraft and with the aid of some excellent colour slides gave his audience their first actual contact with this aircraft, the largest ever built in Canada. The speaker outlined the requirements for building such an aircraft and then explained the functioning of the various committees that were set up to accomplish the task on hand. The relationship between the Bristol Britannia and the Argus were explained in detail, as were many problems involved before the final configuration was approved and accepted. Due to the classified nature of this aircraft and certain of its equipment, Mr. Whiteman was not permitted to give actual figures of performance, but did mention some of the more commonly known electronic systems fitted, the quantity of which was very impressive. In conclusion, Mr. Whiteman dealt with the various phases of both the Company and RCAF qualitative assessment and the different trials and test flying programs involved.

Following his address, a film entitled "Birth of a Giant" was shown. This was a most interesting film dealing with the actual construction of the Argus. Mr. Whiteman was thanked by Mr. H. E. Hampshire on behalf of the Branch for his most informative address.

MEMBERS

NEWS

Lieut. General D. L. Putt, Hon. F.C.A.I., recently retired from the USAF, has been elected President of a new firm, United Research Corp., at Menlo Park, Calif.

Dr. A. Muraszew, A.F.C.A.I., has left Orenda Engines Ltd. to take up a position with Aerojet-General Corp., Azusa, Calif.

H. L. Taylor, A.F.C.A.I., has relinquished his appointment as Assistant Secretary of the Canadian Aeronautical Institute to join the staff of Williams & Wilson Ltd., Ottawa.

F. R. Thurston, A.F.C.A.I., has been appointed Acting Director of the National Aeronautical Establishment, NRC.

W/C R. D. H. Ellis, M.C.A.I., has been transferred from Cold Lake to RCAF Station, Rivers, Man.

W. J. Potocki, M.C.A.I., has been appointed Chief Experimental Pilot of Avro Aircraft Ltd.

G. Rosenthal, M.C.A.I., recently left Beech Aircraft Corp. to take up a position as Aerodynamics Section Head, Republic Aviation Corp., Guided Missiles Division, Mineola, L.I.

J. Zurakowski, M.C.A.I., has retired as Chief Experimental Pilot, Avro Aircraft Ltd., and is now Technical Design, Staff Engineer.

W. Deacon, Technical Member, has terminated his employment with Canadian Aviation Electronics and joined the Engineering Division of Canadair Ltd. as Draftsman in the Aircraft Design Dept.

W. R. Foote, Technical Member, has left Avro Aircraft Ltd. to take up a position as Aircraft Maintenance Engineer with Okanagan Helicopters Ltd.

F/O V. Pullishy, Technical Member, has been transferred from Cold Lake to the RCAF Technical Services Unit at Canadair Ltd.

J. H. McDonald, Associate, formerly senior partner of McDonald, Joyal, Fogarty & Mills, announces the opening of an office for the practice of Law in Ottawa.

ADMISSIONS

At a meeting of the Admissions Committee, held on the 25th November, 1958, the following were admitted to the grades shown.

Associate Fellow

J. A. Chamberlin, Chief of Design, Avro Aircraft Ltd., Box 4004, Terminal A, Toronto, Ont.

W/C C. H. McVeigh, RCAF, Branch Head, Training Devices Branch, AFHQ, Ottawa, Ont.: *1471 Banbury Cresc., Ottawa 1, Ont.*

CDR (E) N. A. Smith (on transfer from Member)

Member

S/L J. P. Bell, RCAF, Accident Inspector, Directorate of Flight Safety, AFHQ, Ottawa, Ont.: *744 Lonsdale Rd., Ottawa, Ont.*

S/L E. A. Bernard, RCAF, Directorate of Scientific Intelligence, Defence Research Board, Ottawa, Ont.: *816 Hamlet Rd., Ottawa 1, Ont.*

S/L D. E. Biden, RCAF, Test Pilot, CEPE/RCAF Stn. Uplands, Ont.: *14 Lipstan Ave., Ottawa 5, Ont.*

W. G. Eves, Industrial Engineering Manager, Avro Aircraft Ltd., Box 4004, Terminal A, Toronto, Ont.

R. Friedman, President, Safe Flight Instrument of Canada Ltd., Toronto Airport, Malton, Ont.

S/L H. E. Holland, RCAF, OC Engineering, CEPE/RCAF Stn. Uplands, Ont.: *1509 Farnsworth Ave., Ottawa, Ont.*

D. M. McAllister, Field Service Engineer, Honeywell Controls Ltd., Toronto, Ont.: *70 Roywood Terrace, Apt. 5, Ville Lemoyne, P.Q.*

S/L W. J. McIndoo, RCAF, Head, Gas Turbine Section, AMC HQ, Ottawa, Ont.: *860 Eastbourne Ave., Ottawa, Ont.*

P. Sefton, Mathematician-Programmer, Computing Devices of Canada Ltd., Ottawa, Ont.: *c/o Officers' Mess, RCAF Stn. Cold Lake, Alta.*

J. H. Washburn, Production Models Engineer, Engines, Canadian Pratt & Whitney Aircraft Co. Ltd., P.O. Box 10, Longueuil, Montreal 23, P.Q.

Technical Member

LAC J. P. G. Berube, RCAF, Electrical Technician, Canadian Armed Forces, Europe: *Damvillers, Meuse, France.*

F/O S. G. Bruskiewich, RCAF, Tech/AE, Officers' Mess, RCAF Stn. Rockcliffe, Ottawa, Ont.

R. J. Gardener (on transfer from Junior Member)

F/O H. J. Moran, RCAF, Project Control, CEPE/RCAF Stn. Rockcliffe, Ont.: *c/o Officers' Mess, RCAF Stn. Rockcliffe, Ottawa, Ont.*

G. D. Morrison, Test Pilot, Canadair Ltd., P.O. Box 6087, Montreal, P.Q.

H. E. Porter, Production Engineer, De Havilland Aircraft of Canada Ltd., Downsview, Ont.: *718 Sheppard Ave. W., Downsview, Ont.*

R. C. Roggeveen, Junior Research Officer, National Research Council, Ottawa, Ont.: *86 Ivy Ave., Ottawa, Ont.*

P. Szego, Design Draftsman, De Havilland Aircraft of Canada Ltd., Downsview, Ont.: *318 Keele St., Apt. 3, Downsview, Ont.*

Junior Member

B. J. Chagnon, Junior Technician, Computing Devices of Canada Ltd., Ottawa, Ont.: *1185 Belmont, Sillery, Quebec City, P.Q.*

L. A. Harvey (on transfer from Student)

Student

G. J. Akamoto, McGill University, Montreal, P.Q.: *3407 Hutchison St., Montreal, P.Q.*

F/C D. L. Klappstein, 4 Sqn. Fort Lasalle, Royal Military College, Kingston, Ont.

Associate

A. D. Belyea, Assistant Director, Aircraft Branch, Dept. of Defence Production, Ottawa, Ont.

S. A. Rutland, Government Contracts Administrator, Servomechanisms (Canada) Ltd., Toronto, Ont.: *1502 Melfa Circle, Ottawa, Ont.*

BOOKS

Nuclear Rocket Propulsion. By R. W. Bussard and R. D. DeLauer. McGraw-Hill Company of Canada Ltd., Toronto, 1958. 370 pages. Illus. $12.00.

If a nuclear reactor replaces combustible fuel as a source of heat energy, the chemical rocket becomes a nuclear rocket. The relevant technologies involving many fields are presented in this encyclopedic book from the viewpoint of the development and design engineer. The coverage has the wealth of material and charts of a handbook, yet it is backed up by a rather remarkable amount of mathematical fundamentals and theory, given clearly, but in the main without derivation.

An early chapter develops the ballistic equations of rockets in general, the thermodynamics of nozzle performance, and propellant characteristics. Another deals largely with problems of vehicle optimization in terms of weight breakdown.

There is an impressive account of heat transfer and fluid flow, covering geometries relevant to reactor heat exchangers. (A sound background in fluid mechanics is presupposed.) This leads into a discussion of materials peculiar to reactors and problems of thermal strain and stress.

The coverage of nucleonics similarly presupposes a basic knowledge of nuclear theory. Reactor statics, kinetics, geometries and the attendant problems of neutron and gamma heating are treated from the design aspect. A survey of the problems of testing and of radiation safety is given.

The chapter on system control devotes several pages to developing the mathematical foundations of the Laplace Transform! The results are applied to feedback-system loops, together with theorems (i.e. Nyquist's) for stability. Also treated are analogue computers for simulating the nuclear rocket system loop and control instrumentation.

The book concludes with a chapter on more "exotic" rocket systems. Consideration is given to use of fusion energy and radioactive decay energy. The all-gaseous fission rocket is discussed at some length. Finally, the fission reactor is visualized as powering an electric generator; this leads on the one hand to the arc-heated jet, and on the other to ion propulsion. The design criteria for ion propulsion are worked out.

With, perhaps, some reservations as to the strict accuracy of points of the fluid dynamics theory (e.g. the constancy of τ and q/A, bottom p 95), the book is highly recommended.

Dr. H. S. Ribner

Gliding, A Handbook on Soaring Flight. By Derek Piggott. Macmillan of Canada Ltd., Toronto, 1958. 261 pages. Illus. $5.00.

Unlike most gliding handbooks, this text has been confined to the practical side of glider pilotage and airmanship from the very elementary stages up to and including aerobatics and advanced soaring. It has been written in three sections, (1) learning to glide, (2) further training and (3) advanced soaring, each section being in logical progression from the previous stage during the training of a glider pilot. Presentation is in clear and simple language and good diagrams have been included to further clarify each subject as it is discussed. Precautions to take and pitfalls to avoid have been covered in detail for both elementary and advanced stages; also particular attention has been given to such facets as the finding and efficient use of thermals, instrument flying, flying at high altitude and effective methods to increase distance covered per flight.

Physically the book is printed clearly and cleanly on good quality paper. Sixteen well produced photos have been included in addition to the seventy-five clearly drawn and reproduced diagrams which are used to amplify the text.

This is a very competently written and up to date addition to the relatively few English language handbooks on the subject. It is highly recommended to all those who are interested in the sport of gliding and soaring.

S/L G. W. E. Brown

APPOINTMENT NOTICES

The facilities of the Journal are offered free of charge to individual members of the Institute seeking new positions and to Sustaining Member companies wishing to give notice of positions vacant. Notices will be published for two consecutive months and will thereafter be discontinued, unless their reinstatement is specifically requested. A Box No., to which enquiries may be addressed (c/o The Secretary), will be assigned to each notice submitted by an individual.

The Institute reserves the right to decline any notice considered unsuitable for this service or temporarily to withhold publication if circumstances so demand.

Positions Vacant

Mechanical or Aeronautical Engineer: A vacancy exists for a graduate mechanical or aeronautical engineer to assist in analysis and investigation on aero engines returned from service. Practical experience in aero engine overhaul shops would be advantageous. Under 30 years of age. Salary subject to negotiation. Apply in writing, giving full particulars of education and experience to Personnel Department, Bristol Aero Engines Ltd., Pie IX Blvd., Montreal North 12, P.Q.

Electrical Engineer: A vacancy exists in the Engineering Office for an Electrical Engineer who should be a graduate of a recognized university and have had at least two years' experience of electrical installation design in military and civil aircraft. Medical and hospitalization benefits. Pension plan after qualifying period. Write giving full details of education, qualifications and experience to the Manager, Industrial Relations Department, Northwest Industries Ltd., Box 517, Edmonton, Alta.

Technical Assistants: For an Engineering Section concerned with the overhaul and repair of gas turbine engines. Work involved will be the examination of engines returned to determine their condition and the compilation of the necessary reports and recommendations. A technical and practical background in aeronautical (engine) engineering is preferable, but not essential. Applicants should be between 22 and 35 years of age. Education to Engineering Standard or equivalent (Higher National Certificate). Salary commensurate with qualifications and ability. Usual employees' benefits. Apply in writing only, giving qualifications and an outline of previous experience, to Assistant Personnel Manager, Rolls-Royce of Canada Ltd., Box 1400, Stn. O, Montreal 9, P.Q.

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FEBRUARY 1959

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The Canadian Aeronautical Institute wishes to announce the forthcoming publication of a special issue of the Journal, which will contain articles of an historical nature written specially for the occasion by the Hon. J. A. D. McCurdy, Mr. K. M. Molson, Mr. J. H. Parkin and W/C H. Pearce.

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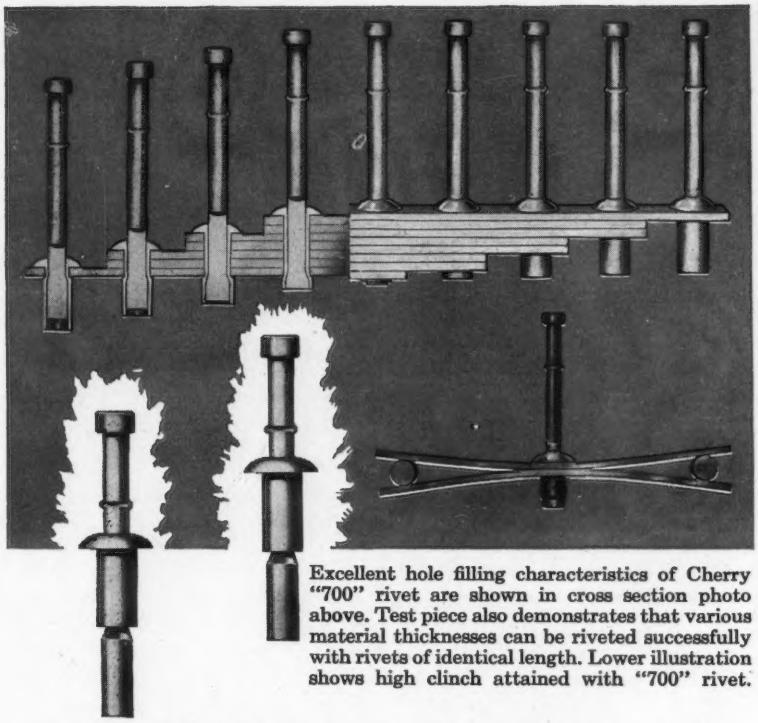

Excellent hole filling characteristics of Cherry "700" rivet are shown in cross section photo above. Test piece also demonstrates that various material thicknesses can be riveted successfully with rivets of identical length. Lower illustration shows high clinch attained with "700" rivet.

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The "700" rivet is available in countersunk and universal head styles in a wide range of diameters and lengths. It is installed with standard Cherry rivet guns with controlled-stroke pulling heads and accessories.

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RELIABLE AND VERSATILE GROUND SUPPORT

Truck installations supply pneumatic power for main engine starting, cooling and heating, electrical power for commercial jet-powered airliners.

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TURBINE-POWERED HELICOPTER

The Bristol Type 192, a twin-turbine, twin-rotor helicopter with complete single engine reliability is now in production.

It follows the versatile Sycamore, a piston engined helicopter which has seen more than eight years service with the Armed Forces.

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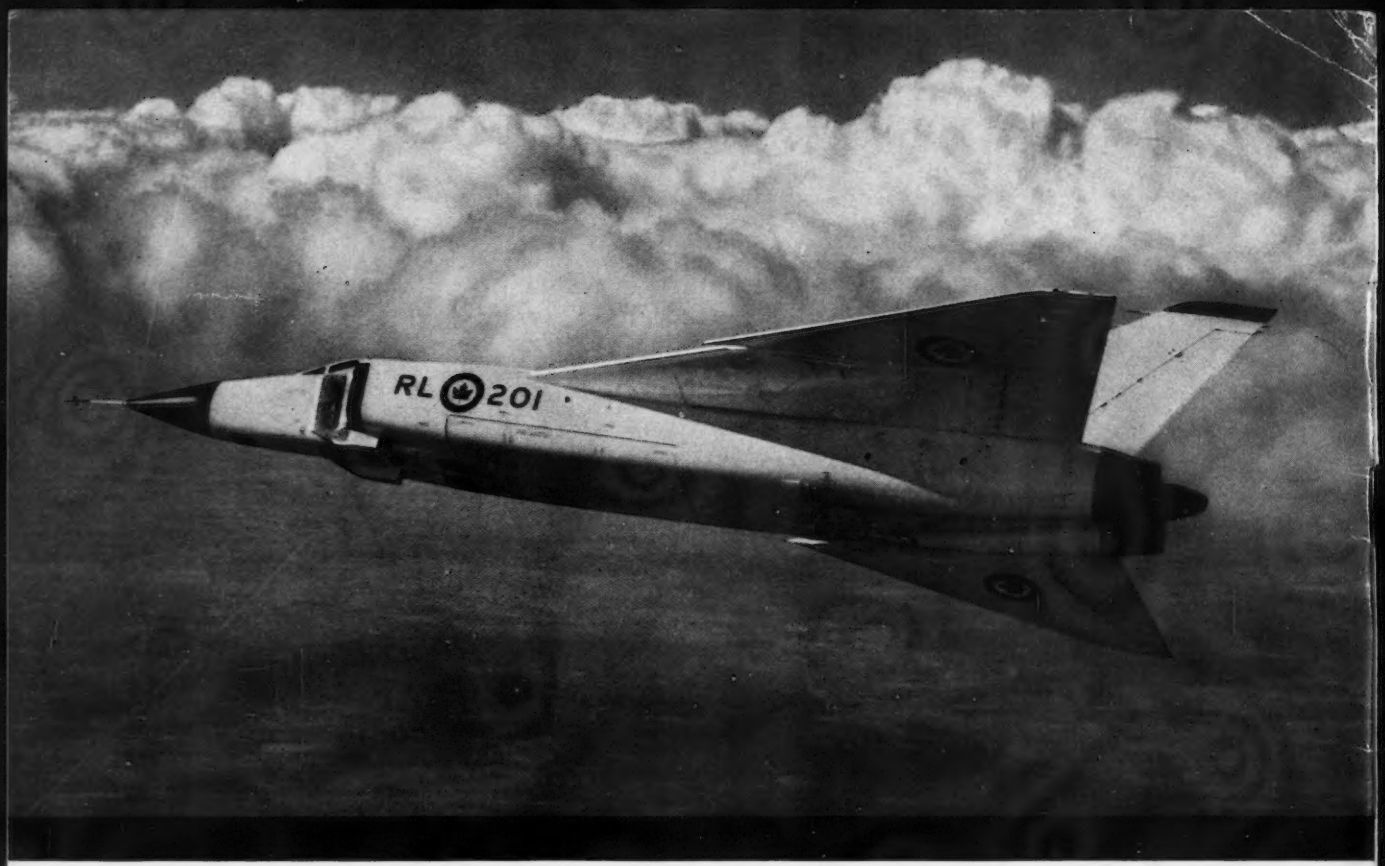
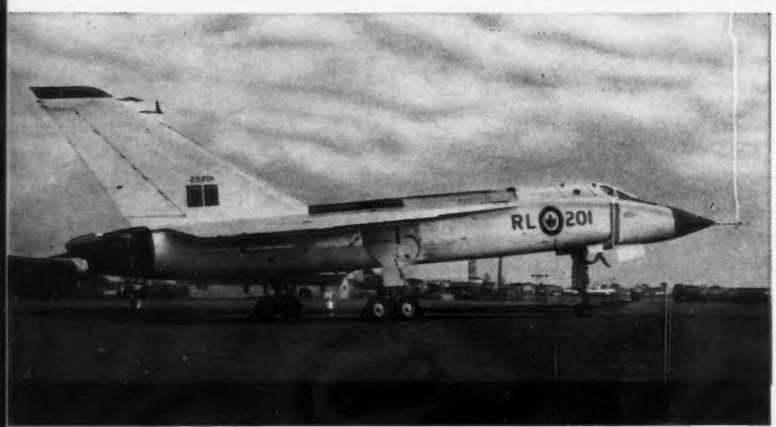
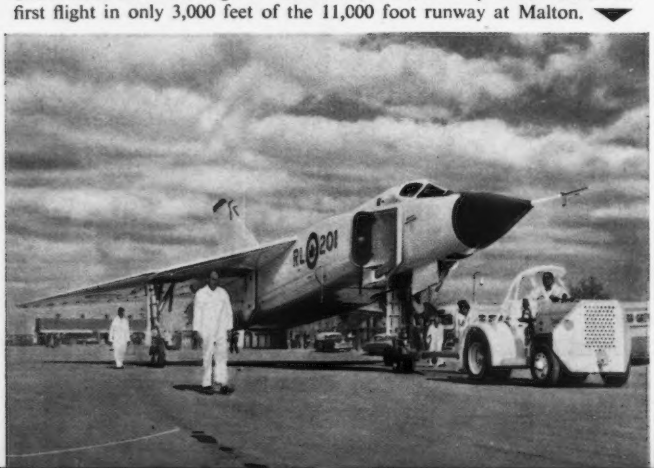


▲ The Avro Arrow is shown in flight during test manoeuvres over Ontario.

The Arrow weapon system is a bomber-destroyer having supersonic mission capabilities. ▼



The Avro Arrow is as big as a World War II bomber yet took off on its first flight in only 3,000 feet of the 11,000 foot runway at Malton. ▼



AN ARROW IN THE SKY

Since its first flight on March 25th, the Avro Arrow has been meeting the vigorous demands of its extensive flight test program. Proceeding according to plan, the Arrow flew faster than sound on its third flight, and more than 1,000 miles per hour on its seventh flight.

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